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QUARTERLY REPORT NO. 1

FEASIBILITY STUDY OF A 200 WATT PER KILOGRAM LIGHTWEIGHT SOLAR ARRAY SYSTEM

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PREPARED FOR

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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ABSTRACT

This report is the first in a series which will describe the technical progress in the investigation of the feasibility of designing a lightweight solar array with a power-to-weight ratio of 200 watts per kilogram. This solar array will produce 10,000 watts of electrical power at 1 A.U. at its beginning of life (BOL), and degrade less than 20% over a three year period in interplanetary flight. A review of existing lightweight solar array system concepts is presented along with discussion pertaining to their applicable technology as it relates to a 200 watt/kilogram array. Also presented is a discussion of the candidate development solar cells being considered, and various deployable boom concepts under investigation.

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SECTION 1

INTRODUCTION AND SUMMARY

A program to study the feasibility of a 10,000 watt solar array subsystem with an overall power-to-weight ratio of 200 watts/kg was initiated on January 19, 1976. This solar array is to be designed for compatibility with a typical 2000 pound interplanetary spacecraft carried aloft by a Space Shuttle. The power-to-weight ratio is interpreted to be the delivered beginning-of-life maximum power output at 1 A.U. and at the predicted array temperature in free space, divided by the total system weight. The total system weight includes the array itself plus all elements of the deployment mechanisms and support structure, but not the gimbaling or orientation related equipment. Thus, for the specified power output of 10,000 watts at 1 A.U., the total system weight must be less than 50 kg (110 pounds). This ultra-lightweight array will require improvements in both the solar cell blanket unit weight and in the elements associated with the deployment and support structure.

The program has been organized into the following major tasks:

Task No.	Task Title
100	Review and Update Technical Requirements
200	Design Synthesis and Analysis
300	Prepare Layout Drawings
400	State of the Art Projections
500	Prepare Subsystem Specifications

The Technical Roadmap to be followed in the performance of this program is shown in Figure 1-1.

In Task 100, the technical requirements imposed on the solar array were reviewed and updated by adding the latest Shuttle launch environments, and the candidate solar cell data provided by JPL. Appendix A includes the baseline requirements to which the 200 watts/kg solar array is being evaluated.

In Task 200, design synthesis and analysis, various existing lightweight array concepts will be analyzed along with new conceptual approaches. Promising system designs will then be selected for further evaluations. Studies will be performed to optimize solar cell blanket size, weight, and aspect ratio. Various deployment boom designs will be analyzed to optimize overall weight as a function of stiffness, tension, and natural frequency. Requirements for solar cell size, contact configuration, interconnecting methods, adhesives, coverglasses, and substrate materials will be determined and fed back into the evaluation cycle. This iterative process will result in an evaluation matrix that compares the relative merits of each of the selected systems. Many other factors will also be considered as part of this Task; such as repairability, reliability, relative cost, testability, handling considerations, structural interface, etc. With the aid of the evaluation matrix, those systems showing the most promise of meeting the overall requirements will be selected for detailed synthesis and analysis. This is intended to verify feasibility of designing a lightweight solar array capable of delivering 10,000 watts of electrical power at 1 A.U. and having a power-to-weight ratio of 200 watts per kilogram.

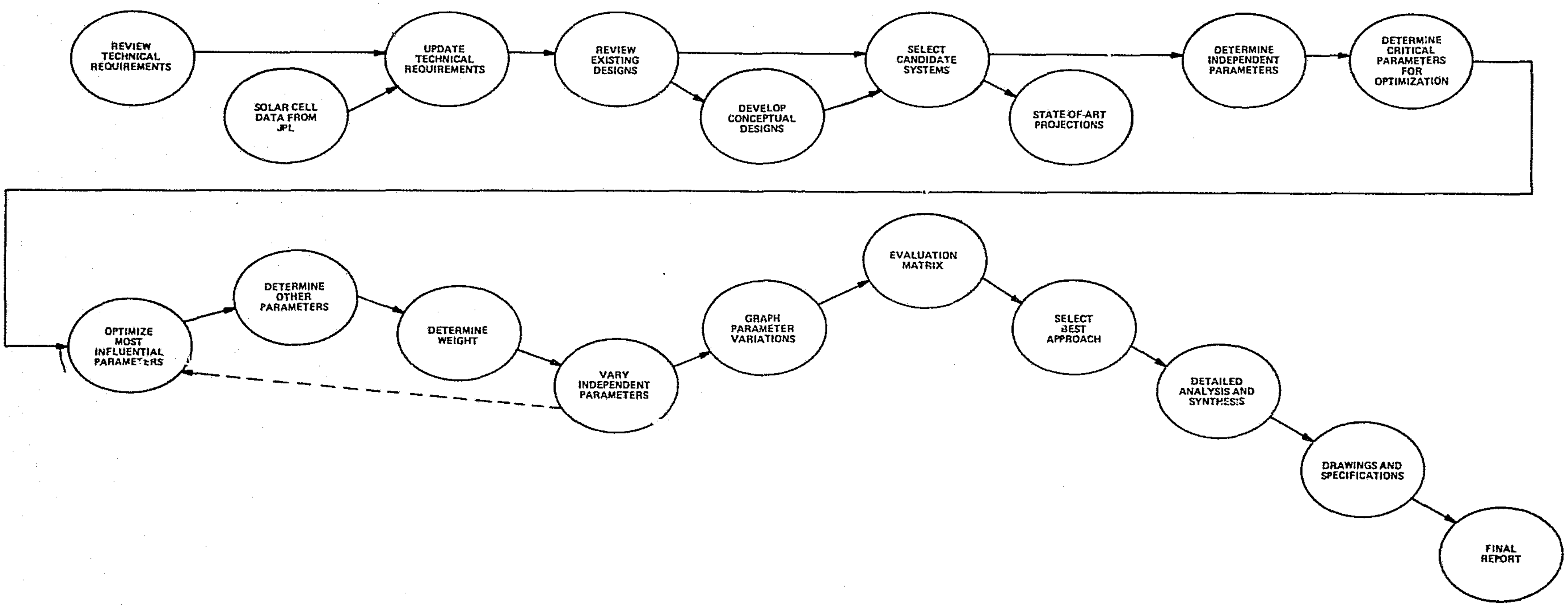


Figure 1-1. Technical Road Map

The results of Task 200 culminate in the documentation requirements under Task 300 (prepare layout drawings), Task 400 (state of the art projections), and Task 500 (subsystem specifications).

The main effort during this first quarter centered around a review of existing lightweight systems and deployment devices. Parametric vibration studies were performed to optimize stiffness, tension, aspect ratio, and natural frequency for planar and V-stiffened arrays. Solar cell data received from JPL was analyzed to determine cell performance as a function of temperature and cell thickness. The overall array dimensions and number of cells required to deliver 10,000 watts at 1 A.U. and at the initially predicted temperature, was also established as a preliminary baseline. Weight summaries for various concepts were also prepared and are discussed in this report.

SECTION 2

TECHNICAL DISCUSSION

2.1 DESIGN REQUIREMENTS

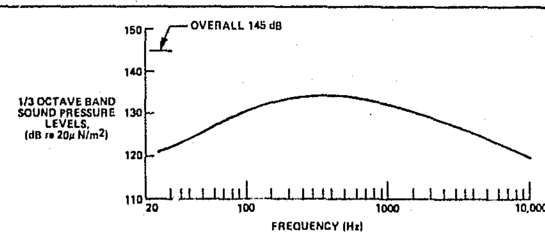
2.1.1 GENERAL

The basic design requirements for the 200 Watt per Kilogram Solar Array Feasibility Study are given in GE document number 200W/Kg - 2.76-004, Baseline Requirements, which is included in Appendix A of this report. In the following paragraphs, the major requirements as they pertain to the solar array will be briefly discussed. Table 2-1 lists these major requirements. These requirements are not intended to place undue restrictions on the solar array system design, but only to act as a guide in the formulation of a design approach. When any design requirement is found to restrict a potentially attractive design approach, this requirement will be reviewed to determine its impact on the ability to achieve the 200 watt/Kg goal. For example, if the deployed natural frequency requirement of 0.04 Hertz is found to impose structural weight penalties on the solar array system which make the 200 Watt/Kg goal unachievable, then that requirement will be treated as a parameter in determining the affect on total system power-to-weight ratio. The intent is to develop high performance design concepts which are viable concepts for future interplanetary missions. These design requirements will be representative rather than specific, as a detailed optimization cycle would be a part of any flight hardware application.

2.1.2 OUTPUT POWER AND DEGRADATION

The output power of the fully deployed solar array is specified as 10KW in free space at 1 AU and at the predicted solar array operating temperature at this intensity. ASTM specification E490-73A defines 1 AU. A copy of this specification is included in this report as Appendix B. The 10 KW power output is defined as

Table 2-1. Major Requirements

Specification Paragraph No. (See Appendix A)	Parameter	Requirement
3.2.2	Electrical Power, Beginning of Life	10 kW in free space at 1 AU and at predicted array temperature at this intensity
3.2.2	Point of Power Measurement	At spacecraft interface
Derived from 3.2.2, 3.2.5	Weight, including deployment mechanisms and all mounting bracketry	50 kg
3.2.5	Power-to-Weight Ratio	200 Watts/kg in free space at 1 AU and at predicted array temperature at this intensity
3.2.3	Operating Lifetime	Not less than 3 years
3.2.3	Power Degradation Limits	No greater than 20% loss of power, excluding solar flare proton reduction, over a period of 3 years
3.2.1	Deployment/Retraction	Full deployment. Retraction up to 90% of exposed array is considered an option
3.2.4	Solar Array Operating Temperature	$85^{\circ}C$ maximum in free space at 1 AU
3.2.8	Deployed Array Dynamics	Lowest natural frequency equal to or greater than 0.04 Hz
3.2.10	Deployed Array Flatness	Lie within predetermined plane with maximum angular deviation of 10 degrees, excluding deflections caused by dynamic load inputs
3.3.2.1	Launch Vibration, **Stowed Array**	Freq. Range (Hz) — Amplitude 2 - 5 — 1.0 Inch Double Amplitude 5 - 26 — 1.3g (0-pk) 26 - 50 — 0.036 Inch Double Amplitude 50 - 1000 — 5g (0-pk)
3.3.2.2	Launch Acoustics, **Stowed Array**	
3.3.2.3	Shock, Stowed Array	20g terminal sawtooth pulse at 10 msec duration at spacecraft interface
3.3.2.4	Static Acceleration, Stowed Array	9g's axial, 2g's lateral
3.3.3.1	Steady State Thermal Vacuum	$-130^{\circ}C$ to $140^{\circ}C$ at 10^{-5} Torr
3.3.3.2	Thermal Shock	$-190^{\circ}C$ to $+140^{\circ}C$ at 10^{-5} Torr, 1000 cycles. Natural heating/cooling rate of simulated passage through planetary shadow (Albedo = 0)
3.4.5	Solar Cells	Use candidate solar cell data provided by JPL. Current-temperature coefficient of 0.03 mA/$^{\circ}$C-Cm^2. Voltage-temperature coefficient of -2.0 mV/$^{\circ}$C. Length = 2 to 4 Cm, width = 2 Cm, Thickness = 0.003 to 0.010 inch. Cell contacts to be silver-palladium-titanium or silver chromium. Interconnect material to be Beryllium-Copper, Kovar, Molybdenum, or Silver. Weldable or solderable. E/I curves per JPL IOM 342-76-D-025, dated 27 Jan 1976
3.5.1.2	Quasi-Static Loads, Deployed Array	1×10^{-3}g maximum

the beginning of life (BOL) value measured at the spacecraft interface. Therefore, losses in solar cell interconnects and cabling (2-3 percent of total power output) must be added to define the array power required. Since the spacecraft power conditioning circuitry will include the necessary diode isolation, no diodes will be used on the array itself.

The solar array must be capable of operating over a three year period with a power degradation not to exceed 20%. Trapped electron and solar flare proton effects are excluded because there is no data presently available for the candidate solar cells being considered. These effects can be added when the data becomes available. The ultraviolet radiation intensity is specified as 1095 days at 2.002 calories /cm^2/ minute. This, along with thermal rejection considerations and overall weight, will be used to establish the type, thickness, and material employed for coverglasses, adhesives, coatings, etc.

2.1.3 POWER TO WEIGHT RATIO

The power-to-weight ratio is specified as 200 watts of electrical power for each kilogram of total system weight. This ratio is based on the 10KW, BOL, power output at 1 AU in free space. This requirement is the main objective of the program. It represents approximately three-times the capability of existing flight arrays. Several approaches are being considered in our attempt to achieve this goal. Weight reductions are necessary in all elements of the array design. This includes lighter cells, coverglasses, substrates, deployment mechanisms, support structures, booms, and other materials and devices. The higher efficiency solar cells presently being considered result in fewer of them being required to

obtain any given output. Cell interconnect and cabling weight may be reduced by designing higher voltage, lower current arrays. Lighter blanket weights result in reduced stiffness requirements to achieve the 0.04 Hertz natural frequency requirement. A reduction in stiffness can reduce the deployment mechanism and boom weights. The structural weight required solely for launch restraint may be left on board the Shuttle or jettisoned once the spacecraft is ejected into interplanetary flight. Some preliminary studies have been performed in these areas and they are described in paragraph 2.5 of this report. The use of a V-stiffened array shows great promise over a planar array with regard to total system weight.[1] Comparisons of these two configurations can also be found in paragraph 2.5 of this report.

2.1.4 TEMPERATURE

There are several temperature requirements imposed on the solar array and these will be discussed in this paragraph.

With reference to Appendix A, paragraph 3.2.4, the deployed solar array will be designed to operate at 1 AU in free space at a steady state temperature not to exceed 85°C. Since the power efficiency of silicon solar cells decreases as temperature increases, the array thermal design should be directed toward a low operating temperature. Previous studies indicate a Kapton substrate with holes cut out at the cell centers, and with a silicone adhesive protective coating applied to the back surface of the cells at the exposed cut-out areas, a steady state temperature of 57.2°C can be achieved.[2] Although this approach of allowing the back surface of the solar cell blanket to radiate to free space is effective in reducing temperature, the cost of implementing this may be significant, since each cell must be covered with an equivalent layer of adhesive to protect against low energy protons. However, this approach and others will be evaluated during

[1] Reference 13, Section 6
[2] Reference 1, Section 6

the course of the program. The thermal evaluations will begin after promising system concepts have been identified.

Paragraph 3.3.3.1 of the Baseline Requirements (Appendix A) specifies the steady state thermal vacuum environment of -130°C to +140°C at 10^{-5} torr. This requirement defines the range of steady state temperatures, at vacuum, over which the solar array shall be capable of operating. It is not required to produce 10KW over this range, but it is required to be fully operational in both the electrical and mechanical modes. (The 10KW of power output is defined as at BOL, 1 AU, and at the predicted array temperature for that intensity). This temperature range effects both the solar cell blanket design and the deployment mechanisms design.

In the solar cell blanket area, materials used for the solar cell "sandwich" must be selected to prevent breakage or damage due to relative differences between their coefficients of thermal expansion. The solar cell sandwich; consisting of substrate, solar cell to substrate adhesive, solar cell, solar cell to coverglass adhesive, and coverglass; must be designed to withstand not only this temperature range, but also the thermal shock values discussed below. Solar cell interconnects must be capable of operating over these temperatures without applying undue stress on the cell contact areas. The induced thermal deformations caused by temperatures cycling between -130°C and +140°C must be kept to values low enough to meet the flatness requirement of 10 degrees specified in paragraph 3.2.10 of the Baseline Requirements.

In the deployment mechanisms and boom area, the temperature range of -130°C to +140°C implies the use of dry lubricants and materials with similar coefficients of thermal expansion, particularly in bearings, bushings, gears, and shafts. If the deployment mechanism is well coupled thermally to the spacecraft structure, the deployment mechanism will operate near relatively benign spacecraft temperatures.

However, if the deployment mechanism is thermally coupled through the bearings of a gimbaling device, the mechanism may operate over a wider temperature range. In this case, it may be thermally driven more by the array temperatures than by the spacecraft temperatures.

The initial deployment from the fully stowed configuration will likely be at a temperature near that of the spacecraft. The main purpose for retraction is to reduce the power output at high solar intensities (nearer to the sun). Thus, retractions will likely be at temperatures in the high end of the range rather than at the low end.

Paragraph 3.3.3.2 of the Baseline Requirements (Appendix A) defines the thermal shock environment. This requirement specifies a temperature range of -190°C to +140°C at 10^{-5} torr. The time rate of change of temperature for cooling shall be the natural cooling rate of the solar panel in a simulated passage into a planetary shadow with an assumed planetary albedo of zero. The time rate of change of temperature for heating shall be the natural heating rate of the solar panel in a simulated passage from a planetary shadow into a normal solar flux of intensity corresponding to a steady state temperature of +140°C on the solar cells. The total thermal shock environment shall consist of 1000 complete heating and cooling cycles.

The thermal shock environment applies to a deployed array and is a survival environment rather than an operational one. This requirement has its greatest impact on the materials selected for the design. When any of these temperature requirements, as well as all other requirements, are found to restrict a potentially attractive design approach, a parametric review will be performed to determine the impact on the ability to achieve the 200 Watt/Kg goal.

2.1.5 DYNAMICS

The Shuttle launch environments specified in paragraphs 3.3.2.1, 3.3.2.2, 3.3.2.3, and 3.3.2.4 define the vibration, acoustics, shock and acceleration values to be considered in the stowed array design. These environments are felt to be relatively "standard" for spacecraft design and, at this early stage of the program, are not unduly driving the array design to the detriment of overall weight penalties. Table 2-1 summarizes these requirements.

The lowest deployed natural frequency shall be equal or or greater than 0.04 Hertz, as specified in paragraph 3.2.8 of the Baseline Requirements. Studies performed using this value and the quasi-static load of $1 \times 10g^{-3}$ showed the limiting element to be the smallest practical size of the deployment booms evaluated[3]. These studies are described in paragraph 2.5 of this report. Again, at this early stage of the program, neither the natural frequency or the quasi-static load appear to unduly limit the design approaches. However, as the Study Program progresses, it may be necessary to re-evaluate these and other requirements if they are found limiting from a performance standpoint.

2.1.6 SOLAR CELLS

Typical candidate development solar cell data was provided by JPL for incorporation into this Study Program. Paragraph 3.4.5 of the Baseline Requirements (Appendix A) describe this data. Options regarding cell size, thickness, contact configuration and material, interconnecting materials, and grid line density were given. These options will be evaluated to establish optimum cell design for this program. Performance curves of cell output power as a function of thickness, temperature, the number required, etc., are discussed in paragraph 2.4.2 of this report.

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[3]Reference 13, Section 6

2.2 EXISTING SOLAR ARRAY CONCEPTS

2.2.1 GENERAL

A number of lightweight solar array system concepts have been developed with experimental hardware and testing applied to many of the designs. The most recent example of large lightweight arrays in the 25 kw system currently being developed by LMSC for the Solar Electric Propulsion Spacecraft (SEPS).

These concepts can be categorized by three basic approaches, namely, roll-up, flat pack, and folding panels. Roll-up designs require a complete flexible blanket type of array. Flat packs usually involve a series of either flexible or semi-rigid panels which fold accordian fashion into a lightweight container. Folding panels are rigid in construction utilizing a frame or structural material such as honeycomb for a substrate. The frames are large and hinged to form two or more layers when stowed. A comparison summary of efficiencies and power levels of many of these concepts is shown in Table 2-2.

2.2.2 GE/JPL 110W/KG ARRAY

A study of the design of a 110 watt/kg system was completed by General Electric, Space Division in May of 1973 and reported on in Reference 1 of this report. Since this effort represents a starting point from which one can proceed to higher efficiency levels, a description of that baseline design as presented in this reference follows.

The baseline configuration which meets those requirements for the interplanetary and geosynchronous missions is shown in Figure 2-1. This concept consists of a single, central, deployable mast which supports two flexible solar cell blankets. The 10,000 watt beginning-of-life output is generated

Table 2-2 Summary of Existing Light Weight Solar Designs

Type	Efficiency Watts/kg	Efficiency Watts/M^2	Power Level KW	Description Array/Support/Stowage	Source	Remarks
Roll up	77.6	118	2.75	Blanket/Pantograph/Graphite Epoxy Drum	Fairchild-Hiller	
	71.7	118	2.75	Blanket/BI-STEM/Beryllium Drum	General Electric	Eng'g Model Tested
	68.1	118	2.75	Blanket/Ryan Boom/Magnesium Drum	Ryan	
	48.5	107	1.5	Blanket/BI-STEM/Magnesium Drum	Hughes	1971 Flight Experiment
	18.5	118	.08	Blanket/STEM/Drum	British Aircraft Corp.	
Flat Pack	110	95	10	Blanket/Astro Mast/Be Container	General Electric	Study
	83	118	2.75	Panels/BI-STEM/Container	TRW	
	79	118	14.2	Panels/Inflatable Framework/Magnesium Frame	Messer-Schmitt	
	78	118	2.7	Panels/BI-STEM/Frame Attached to S/C	LMSC	
	75	118	.75	Rigid Panel/Telescopic Mast/Container	RAE	
	66	99	25 (2 panels)	Blanket/Astro Mast/Honeycomb Container	LMSC	SEPS Array
Folding Panel	80	90	10.4	Hollow Core Substrate/-/Beryllium Box Frame	EOS	
	40	118	15.7	Fiberglass Be Frames/-/-	Boeing	

by 226,800 solar cells which are interconnected to supply power at a 193 vdc maximum power voltage. These solar cells are nominal 125 μm thick, 2 x 2 cm, N/P silicon with a nominal base resistivity of 10 ohm-cm. A plated nickel-copper-nickel-gold bottom wraparound contact configuration is used in conjunction with an ultrasonically bonded aluminum wire interconnector system. The active solar cell surface is protected from low energy proton damage by a nominal 37 μm thick integrally deposited coverglass. A Kapton-H film substrate supports the solar cell modules without the aid of a bonding adhesive. Holes in the substrate allow for this direct radiation heat transfer from the rear of the solar cells. The exposed portion of the rear cell contacts are coated with adhesive to provide the necessary low energy proton protection.

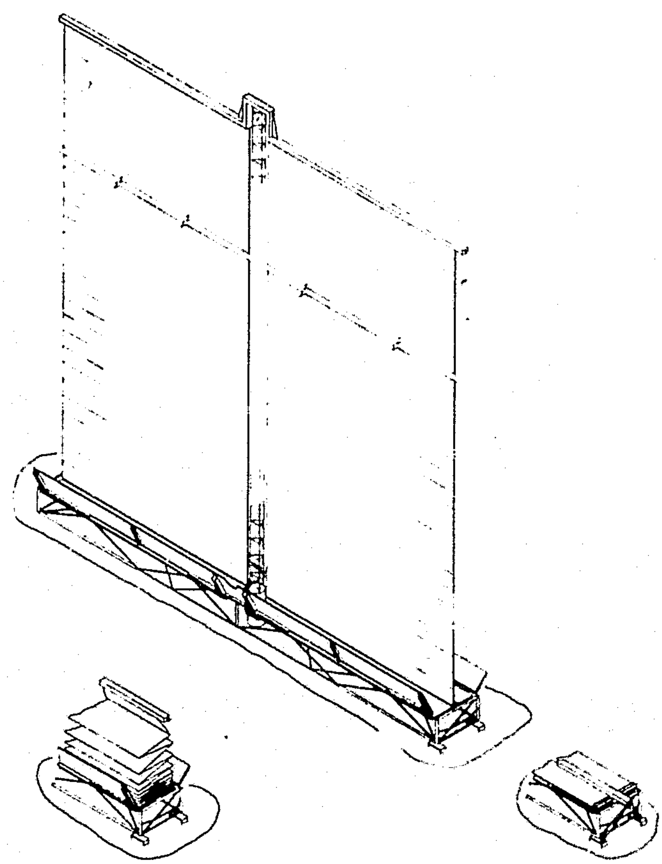


Figure 2-1. Baseline 110W/kg Solar Array Configuration

Tension in the solar cell blanket substrates maintains the deployed natural frequency above the minimum specified value of 0.04 Hz. The flexible solar cell blankets are stowed for launch by folding into a flat-pack package which is retained in compression between a bottom honeycomb panel and spring driven hinged honeycomb panel doors on the top. These doors are held closed during launch by the tubular leading edge member (LEM) which is attached to the deployable boom at the center and retained at each end by a launch retention cable mechanism. Solar array deployment is accomplished by firing redundant cable cutters at each end of the array which releases the end of the LEM and the restraint at each end of the supporting truss work. Application of power to the deployable boom actuator will cause the LEM to move off the door panels allowing them to swing open. Continued deployment of the boom will cause the LEM to pull each fold of the blankets from the stowed package. Interlayer cushions of Kapton-H film are retained by the bottom panel. At the end of the deployment travel, the further deployment of the blanket applies the required tension load by extending a spring mechanism at the base of each blanket.

The deployable boom is an ASTROMAST structure manufactured by Astro Research Corporation (formerly SPAR Aerospace Products, Ltd.). For this interplanetary mission application an articulated longeron mast, which is similar to the Lockheed Space Station Solar Array Mast, may be required to meet the specified +140°C upper temperature extreme.

Some significant design features of the baseline configuration which meets the 110 watt/kg goal are presented in Table 2-3.

Table 2-3, Significant Design Features of the 110 W/kg Baseline Solar Array Panel Configuration

Parameter	Value
1. Deployed length (L')	18.565 m
2. Total width (W)	5.915 m
3. Blanket width (w)	2.830 m
4. System aspect ratio (L'/W)	3.14
5. Total gross blanket area	105.08 m^2
6. Total number of solar cells (2 x 2 cm)	226,800
7. Lowest deployed natural frequency	0.04 Hz
8. Electrical power output at V_{mp} = 193 vdc • Beginning-of-life (BOL) • 57°C • 1 A.U., AMO illumination • Measured at panel interface connector	9860 watts
9. Expected maximum power degradation after 3 year interplanetary mission	30%
10. Total system mass	87.5 kg
11. BOL power-to-mass ratio	112.7 watt/kg
12. EOM power-to-mass ratio	78.9 watt/kg

Each solar cell blanket consists of an interconnection of 30 identical strips as shown in Figure 2-2. Each strip consists of two series connected solar cell modules, with each module being composed of 1890 2 x 2 cm solar cells which are interconnected 135 in series by 14 in parallel as shown in Figure 2-2. The two modules on one strip are connected in electrical series with the two modules on an adjacent strip to form a complete electrical circuit. Thus, each electrical circuit is composed of 7560 cells connected 540 in series by 14 in parallel.

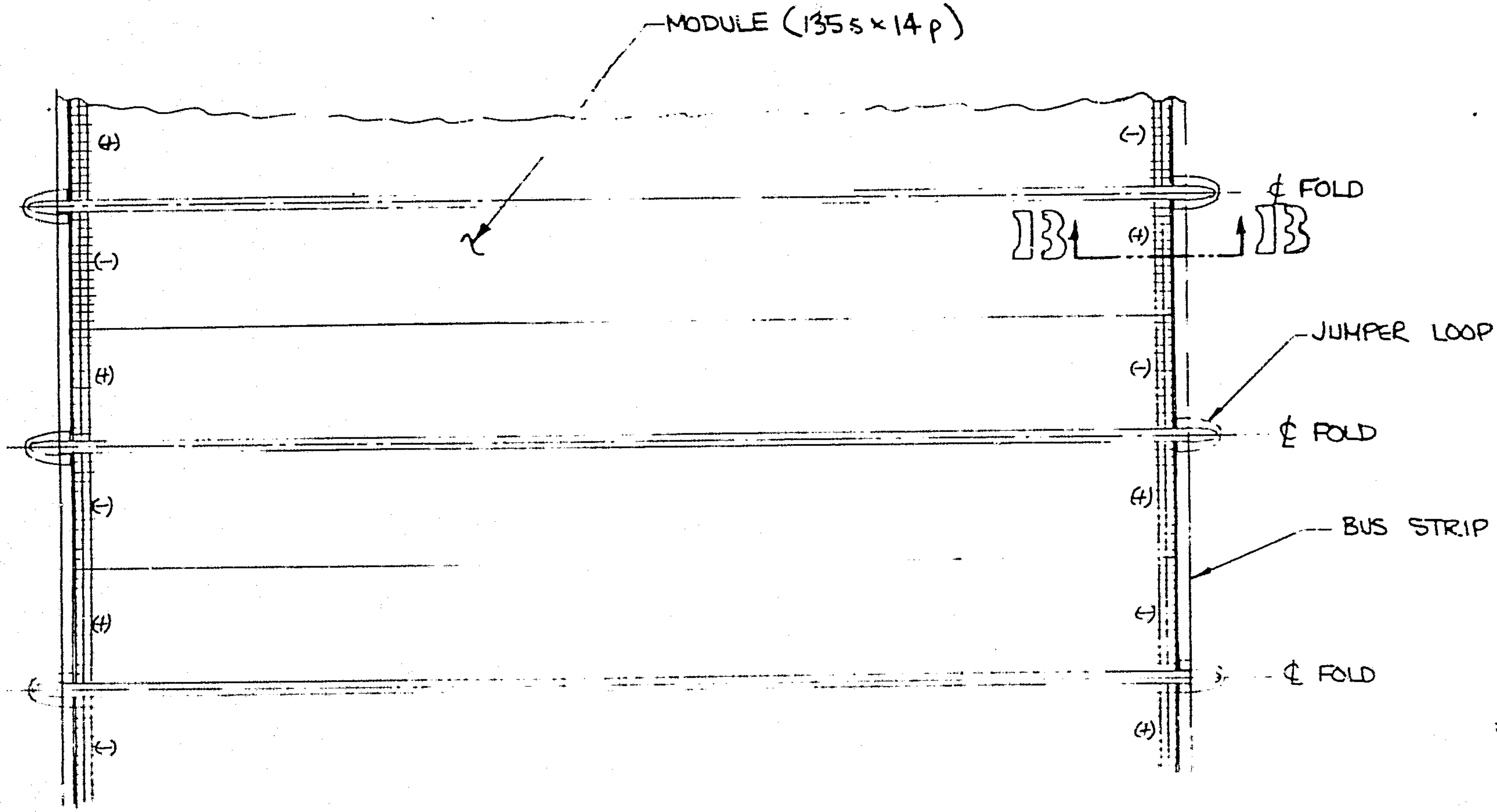


Figure 2-2 Detail of Module Arrangement on 110 W/kg Solar Cell Blankets

Each circuit has a calculated 335 watt maximum power output at 196 vdc measured at the circuit terminals. If a 2-percent bus strip distribution loss is accounted for, the total calculated panel output is 9860 watts measured at the panel interface connector. Table 2-4 is a summary of the component quantities as related to the level of assembly of the panel.

Table 2-4 Component Quantity Related to Level of Assembly - 110 W/kg

	Cell	Module	Strip	Circuit	Blanket	Panel
Cell	1					
Module	1890	1				
Strip	3780	2	1			
Circuit	7560	4	2	1		
Blanket	113,400	60	30	15	1	
Panel	226,800	120	60	30	2	1

The solar cells are nominal 125 μm thick, 2 x 2 cm, N/P silicon with a nominal base resistivity of 10 ohm-cm. Table 2-5 summarizes the significant characteristics of this cell. The solar cells are shielded from the damaging effects of low energy protons by the deposition of an integral cover of Corning 7070 glass. A nominal integral coverglass thickness of 37 μm should provide the necessary protection within the weight constraints of this program.

Table 2-5 Design Characteristics of Ferranti 125 μm Thick Solar Cells (Ferranti Cell Type MS36)

Feature	Description
Thickness	125 ± 25 μm
Size	20 ± 0.15 x 20 ± 0.15 mm
Resistivity	7 to 12 ohm-cm Float zone silicon
Contact Configuration	Bottom wrap-around 24 finger grid geometry
Contact Material	Plated - nickel, copper, nickel, gold
Anti-reflective Coating	TiO_x
Minimum Lot Average Electrical Performance (covered)	123 ma at 0.445 volts (equivalent AMO, 1 A.U. illumination at 25 ± 2°C)
Maximum Lot Average Cell Mass	0.129 gm/cell

Figure 2-3 shows an enlarged rear view of the substrate. The aluminum wire interconnectors are ultrasonically bonded to the gold solar cell contacts through slotted holes in the Kapton film. The wire is pinched flat in the bond areas to allow several bonds to be made at the one attachment point. Between these attachment points, the wire follows a curved path to accommodate the differential expansions and contractions which occur when the array is thermal cycled between -190°C and +140°C. Holes have been cut in the Kapton-H film substrate to allow the rear of the solar cells to radiate directly to space. The rear of the solar cell which is under the hole is coated with Dow Corning 93-500 adhesive to provide the necessary low energy proton protection. To further reduce the solar cell temperature, a high emissivity coating is applied over the entire rear solar cell contact surface except at the points of interconnector attachment.

The bus strips, which run on the sun side of the substrate along each edge, consist of flat copper conductors with Kapton-H film used as the insulator. The electrical connections between blanket strips are made by jumper loops which attach one bus strip segment to the adjacent bus strip segment. The installation of these jumper loops is shown in Figure 2-3. The fold hinge between blanket strips consists of a strip of FEP-Teflon which is heat sealed to the Kapton substrate to form a lap joint along the width of the blanket.

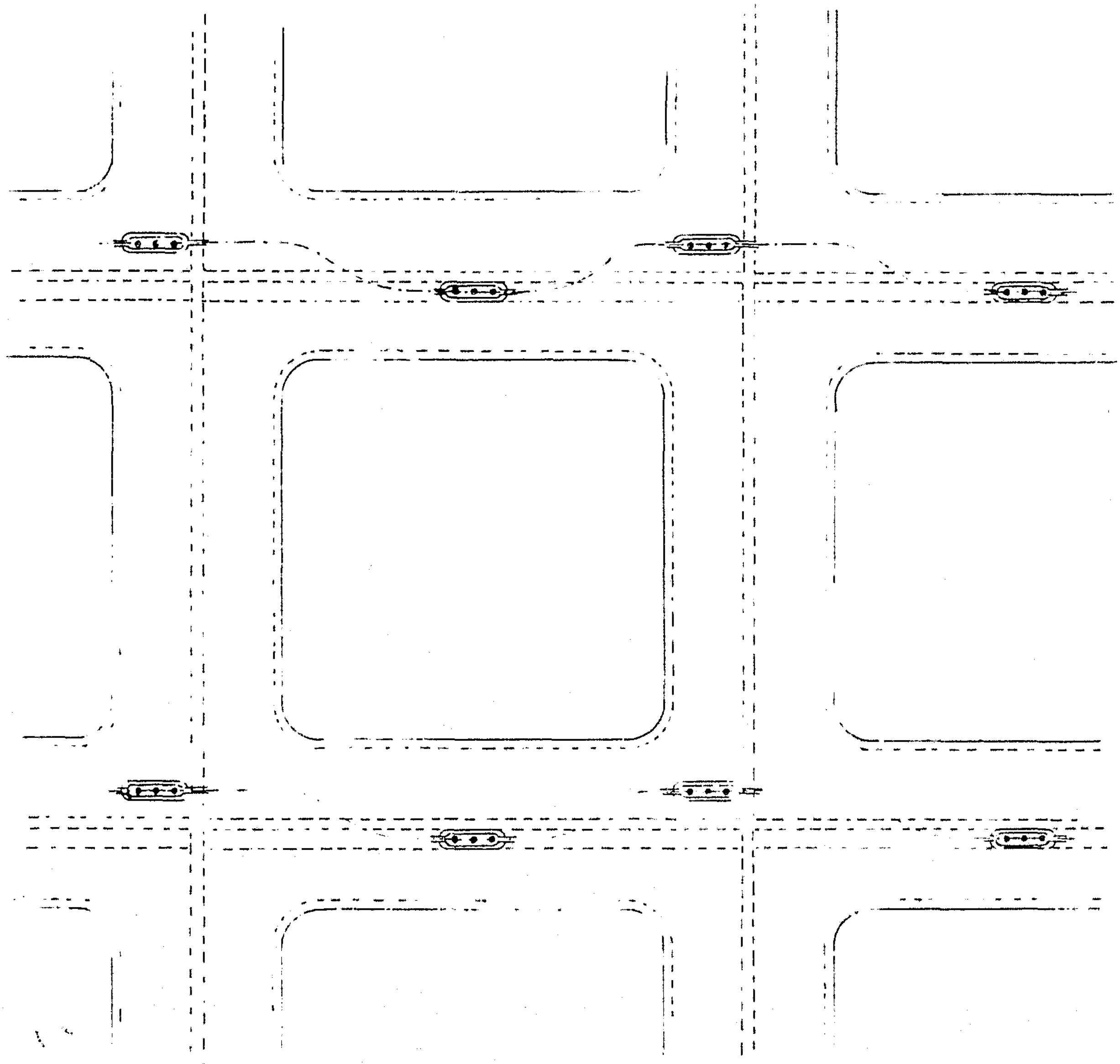


Figure 2-3 Rear View of Solar Cell Blanket for 110 W/kg Array

A detailed mass breakdown for the solar cell blanket is given in Table 2-6.

Table 2-6 Mass Breakdown for Solar Cell Blanket - 110 W/kg Array (Total for Both Blankets)

Items	Mass (kg)
Solar Cells (.129 gm each)	29.26
Integral Coverglass	7.26
Interconnectors	1.81
Substrate	4.04
Adhesive (rear contact low energy proton protection)	3.50
Bus Strips and Insulators	1.50
Inboard and Outboard Leaders	0.27
Circuit Terminations	0.15
Strip Hinge Joints and Bus Strip Jumpers	0.71
	48.50

2.2.3 LMSC SEPS ARRAY

The SEPS array currently under development by LMSC is an extremely large array consisting of two wings with a total output of 25 KW. The baseline configuration is shown in Figure 2-4.

This array is a blanket fold up type using a continuous longeron lattice lightweight extendible mast. The array is stored in a supporting container when retracted. The system is designed to meet the basic requirements listed in Table 2-7.

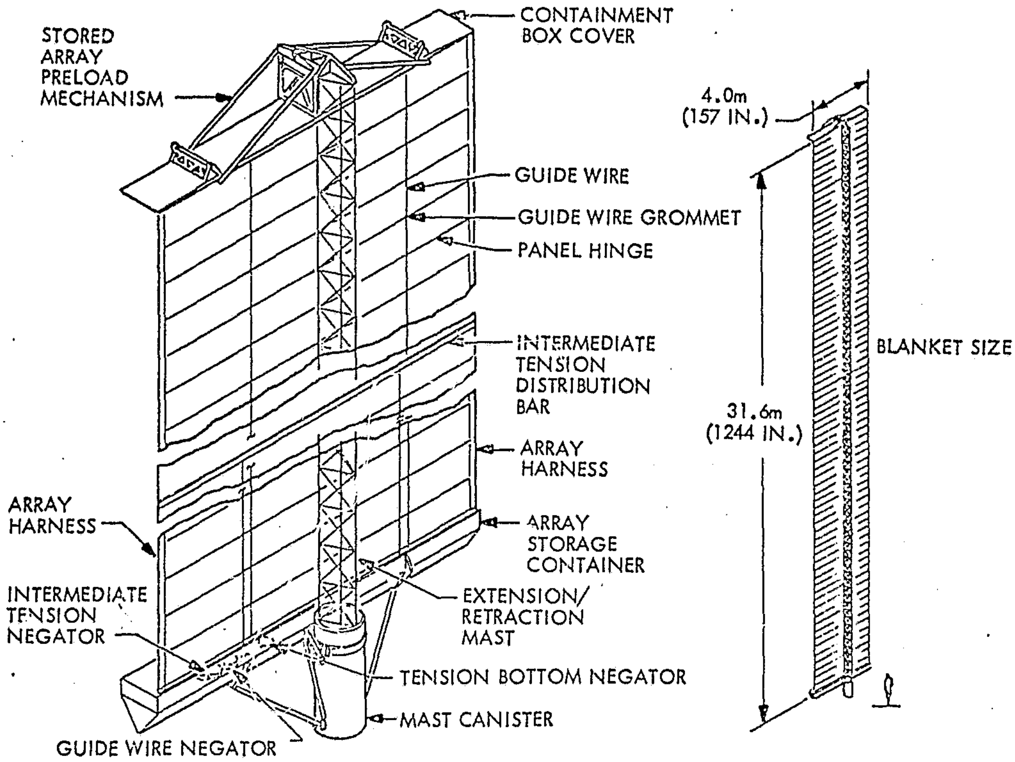


Figure 2-4 SEPS Baseline Configuration

- GOOD JOINT ACCESS
- HIGHER PACKING FACTOR
- REDUCED ASSEMBLY STEPS AND COST
- ELIMINATES N-BUS GAP PROBLEM
- REDUCED WEIGHT

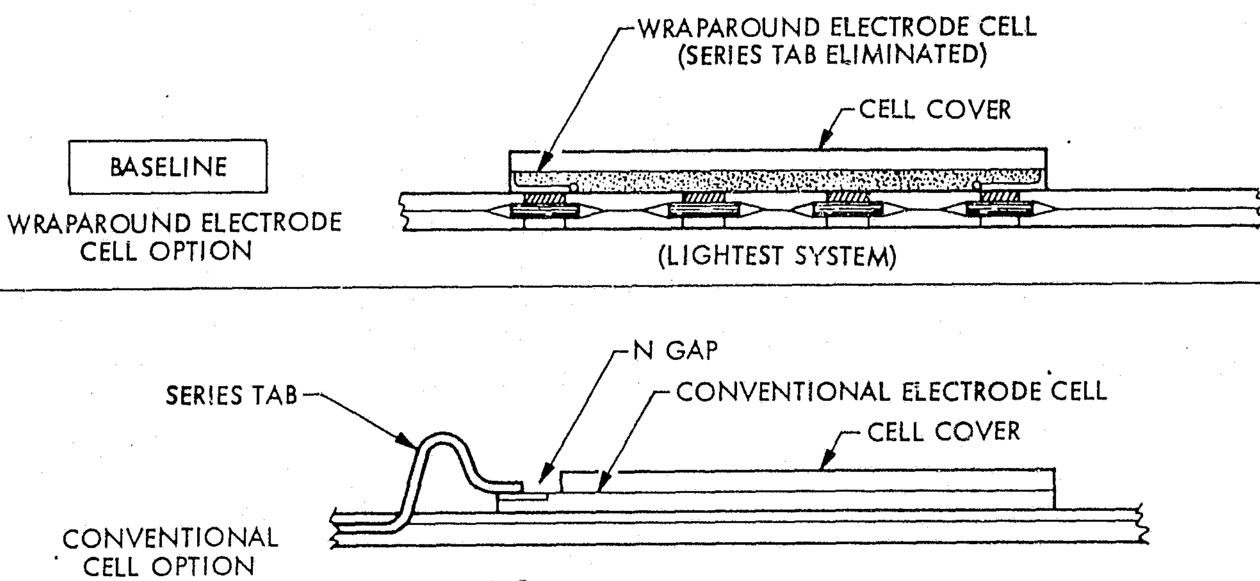


Figure 2-5 Integral Interconnects - SEPS Substrate Design

Table 2-7 SEPS Array Basic Requirements

Parameter	Requirement
Deployment	Full development and retractions from intermediate positions
Stowage Volume	0.46 M x 0.46 M x 4.06 M
Weight	380 Kg maximum
Deployed Natural Frequency (1st mode)	0.04 Hz
Docking Loads	0.5 g
Solar Operating Range	Between 0.3 and 6.0 A.U. from sun
Array Power	25 KW BOL 21 KW EOL
Voltage	V_{oc} 420 VDC V_{mp} 200 VDC
Mast Length (extendible)	32.0 M (105 ft)
Mast Bending Stiffness	5.45×10^4 N-M^2 (19.1×10^6 lb-in^2)
Mast Life	50 cycles full as partial extensions over 5 yr. period in space. 200 cycles ground test

The solar cell blanket is 30.99 M (101.6 ft) X 3.99 M (13.09 ft) in size and is made up of individual panels which are attached to each other by means of a fiberglass and graphite/epoxy piano hinge. When retracting the blanket, the panels are guided by means of wires which are maintained in tension by negator spring motors. The stowage container cover also serves as an outboard header for the blanket array.

The array is comprised by 8 mil thick solar cells with 6 mil covers. The weight density distribution of the baseline design is shown in Table 2-8. A cross sectional view of the cell/blanket construction is shown in Figure 2-5 with the benefits obtained with the wrap around cell electrode.

Table 2-8 Baseline Weight-Density, SEPS Array

Item	Density (lbs/ft^2)
8 mil cell	0.0831
6 mil cover	0.0569
cover adhesive	0.0092
1 mil Kapton	0.0074
1 mil adhesive	0.0073
1 mil copper	0.0116
	0.1755 Total

2.2.4 GE/JPL 30 WATT/POUND SOLAR ARRAY

In February 1971 General Electric completed a program to develop the technology of the rollup solar array concept under contract to the Jet Propulsion Laboratory. This array, shown in Figures 2-6 and 2-7, provides 2.75 KW at an efficiency of 71 watts/kg. A flexible Kapton substrate blanket 1.17 m (46 inches) wide by 10.2 m (402 inches) long is stowed by rolling it up on an 8" diameter drum of sheet beryllium. A negator spring motor on the drum provides the blanket tension. The array is deployed and retracted by means of an Astro

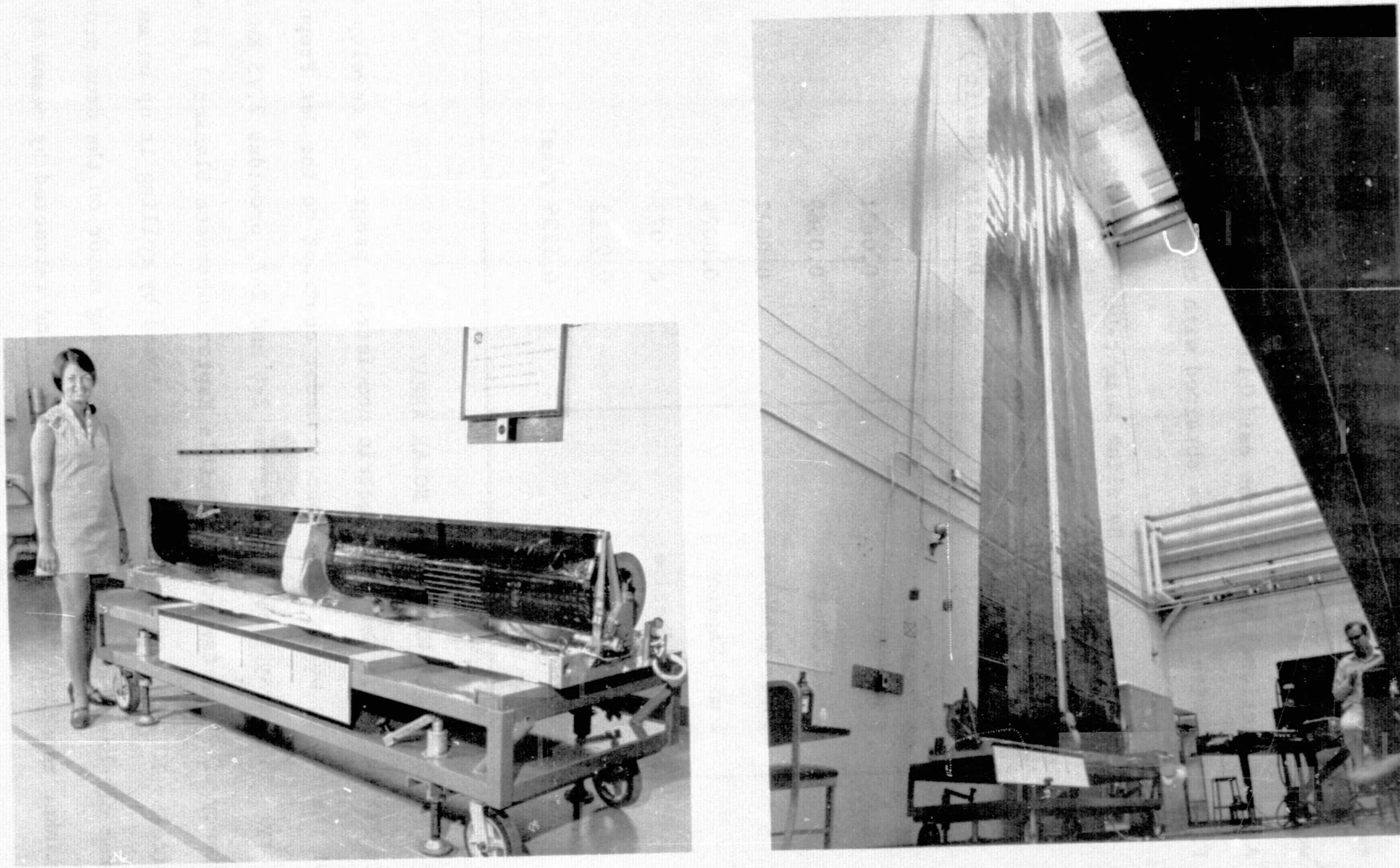
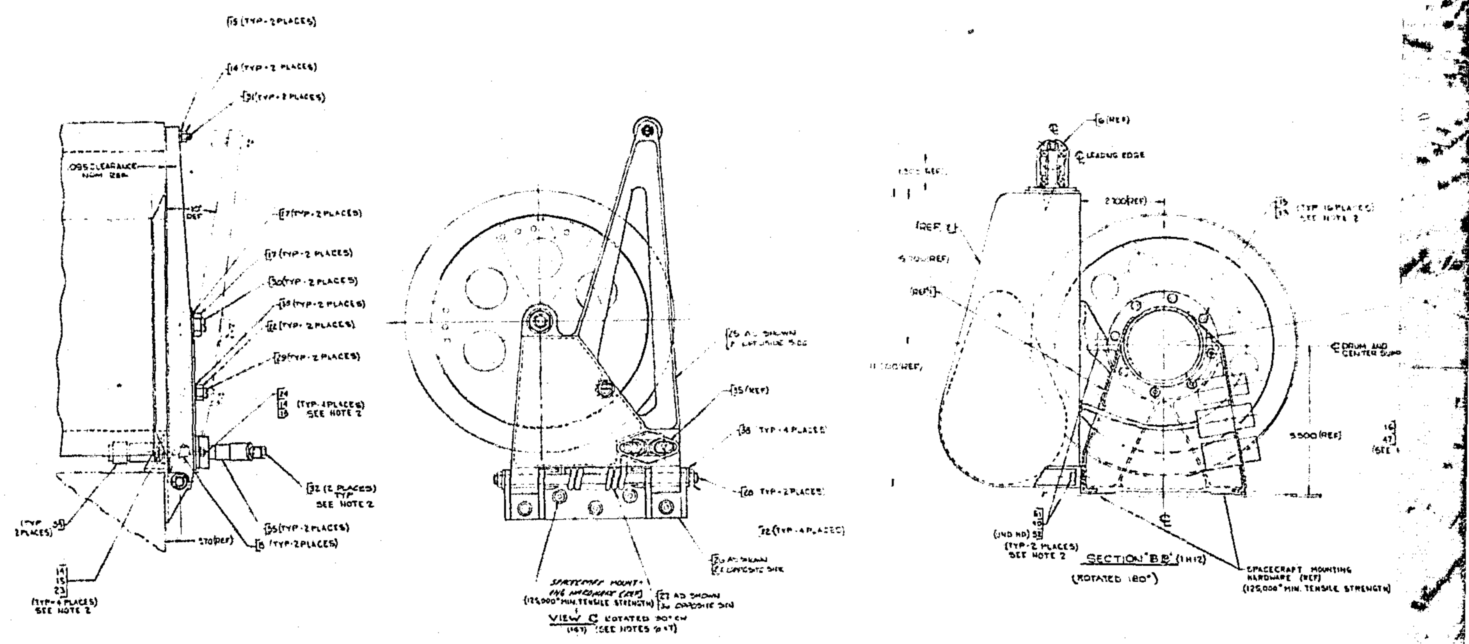
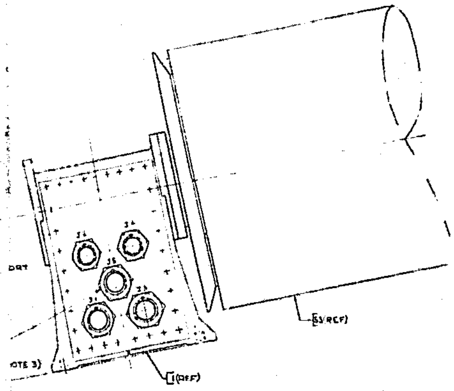


Figure 2-6 RA250 Prototype Test Model

FOLDOUT FRAME 1





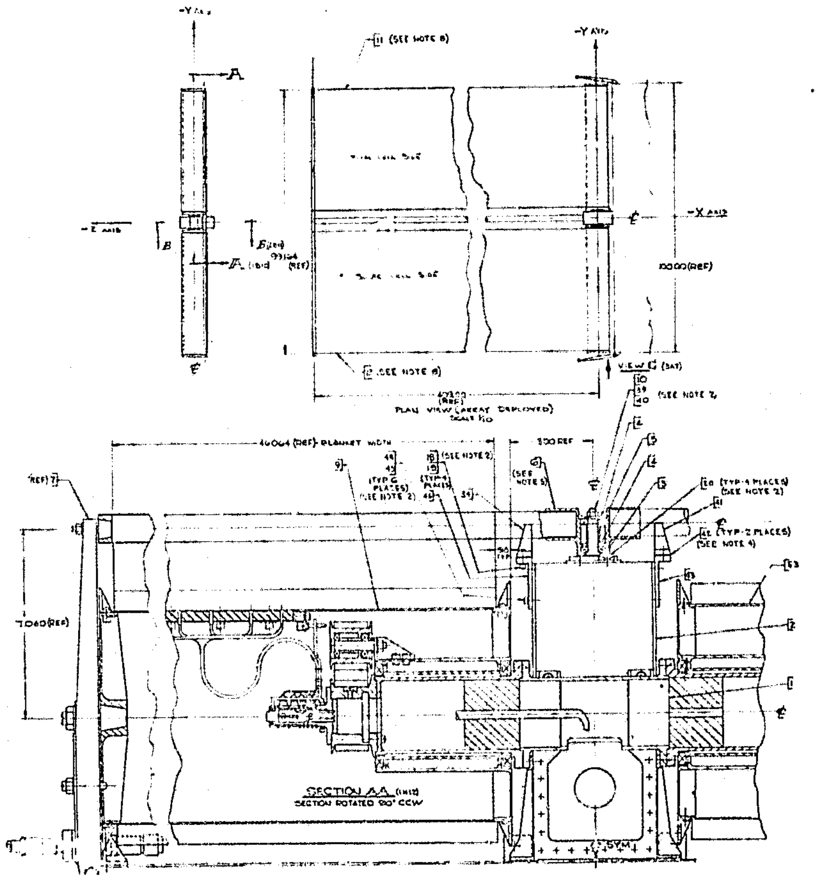
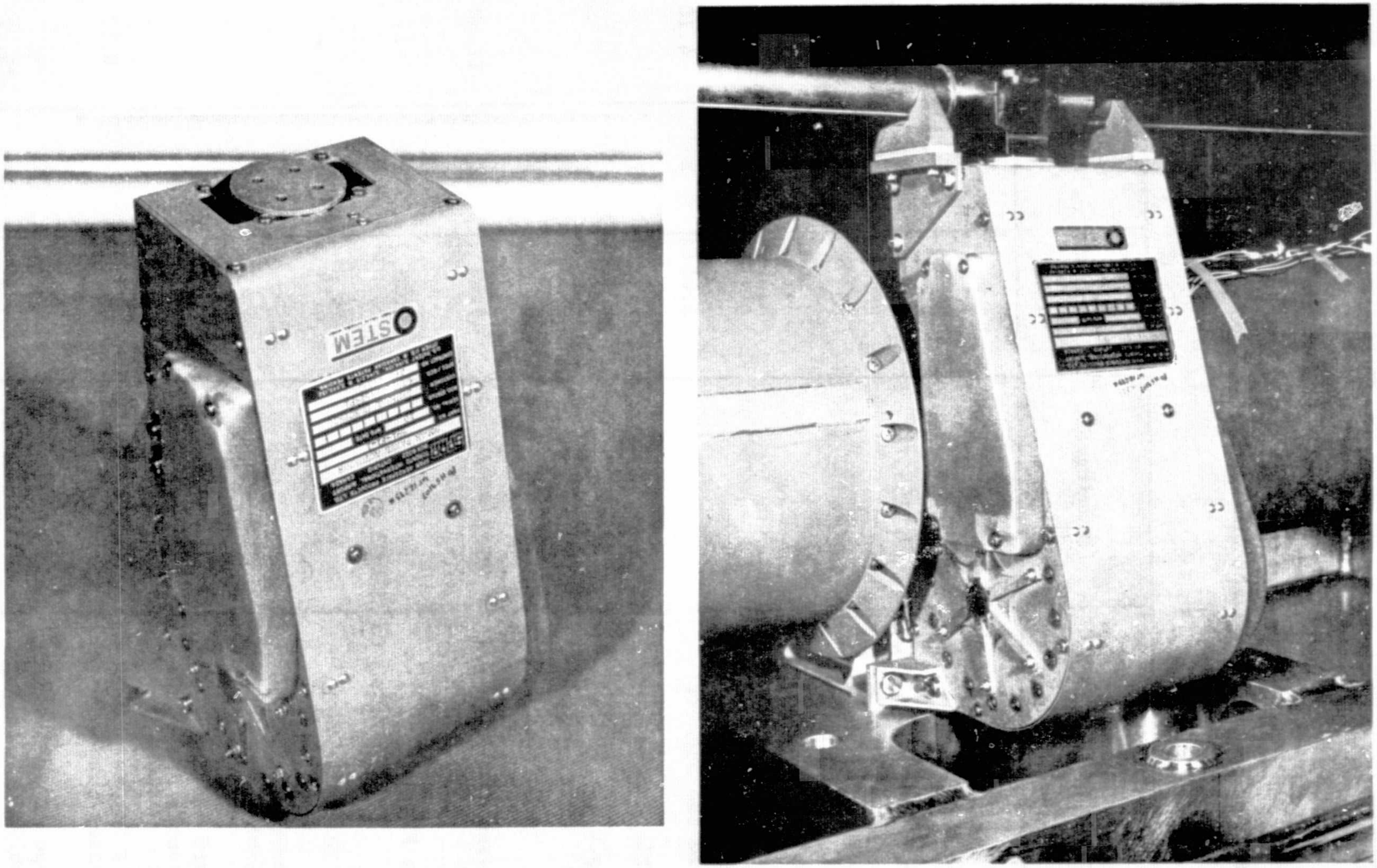


Figure 2-7 RA250 Assembly

Research "BI-STEM" deployable boom 1.34" in diameter shown in Figure 2-8.

The array was fabricated with partial solar cell coverage and the remaining area with dummy glass modules. Cell modules were procured from several suppliers to provide a representative sampling of interconnection approaches, including the resources of Heliotek, Spectrolab, Boeing, EOS, and Centralab. Slip rings conducted power from the array to electrical interface connectors. The total weight of the array was 37.4 kg (82.5 lbs.). A summary weight distribution is presented in Table 2-9. A prototype model was fabricated and tested for performance and response to the environments of vibration, pyrotechnic shock and thermal vacuum. The prototype array demonstrated that a 250 sq. ft. array could be made to exceed the baseline target of 30 watts per pound. It provided valuable information on such items as cell module variations in weight and construction, rollup extension and retraction, dynamic response, damping characteristics, and thermal radiation effects. Detailed results of this effort are found in Reference 2.



(a) Bi-Stem Component (VF70136D)

(b) Mounted on Center Support (VF70136C)

Figure 2-8 Solar Panel Actuator

Table 2-9 Actual Weight Summary (Prototype Model)

	Nomenclature	Drawing No.	Unit Weight (lb)	Qty/ Next Assy	Total Weight (lb)
RA250	Prototype Assembly	47E214519G2	-	-	82.5
	Center Support	47E218547	1.33	1	1.33
	Leading Edge Member	-	0.85	1	0.85
	Boom Actuator	-	11.73	1	11.73
	LEM Support Brackets	-	0.11	2	0.22
	Outboard End Support	-	2.05	2	4.10
	Movable Portion	-	1.31	1	-
	Fixed Portion	-	0.69	1	-
	Bolt	-	0.05	1	-
	Drum Assembly	47E218804 G3 & G4	8.80	2	17.60
	Guide Flange	47D218535 G3 & G4	0.38	2	-
	Drum Shell	47E218144G4	2.79	1	-
	Outboard End Cap Assembly	47E218194G3	0.45	1	-
	Inboard End Cap Assembly	47E218544 G1 & G2	4.80	1	-
	Mounting Hardware (Drum-to-Center Support)	-	-	-	0.13
	Prototype Array Blanket Assembly	475218819G1	23.22	1	23.22
	Prototype Array Blanket Assembly	47J218819G2	23.36	1	23.36

2.3 ADVANCED SOLAR ARRAYS

A solar array system which can produce 200 watts/kg of overall weight represents a significant design challenge. As noted in the summary chart Table 2-2, most arrays developed to date have been in the range of 65-75 watts/kg, in respect to power versus weight performance. One intent of this conceptual study is to keep abreast of all current development activities in the various areas of blanket fabrication, extendible structures, and stowage techniques, so that maximum benefit may be derived from the results of these programs. Work being performed at LMSC for NASA on flexible solar array design optimizations, and on the development of SEPS flat pack approach to large arrays, is of particular interest.

2.3.1 SOLAR ARRAY SYSTEM TRADES

In the development of a solar array system design, a number of design decisions must be made which will be governed by considerations of performance, minimum weight, reliability in space, and producibility. Some of the items which will receive consideration in this study are discussed herein.

2.3.1.1 Array Configuration

The shape of arrays may be made rectangular, square, or round. Most arrays to date have been rectangular, in which case a choice of aspect ratio is necessary. Long narrow arrays may pose a problem of too low a natural frequency which interferes with spacecraft attitude control stability. It is usually desirable to make an array relatively short normal to its axis of rotation so that orientation drive reaction torques are minimized. Round shapes are not efficient in generating cell mounting area and hence would have to be justified on the basis of associated weight or ease in retraction and deployment.

The number of arrays (wings) in the system is often a decision for the spacecraft system designer. Breaking the system up into separate wings can reduce the size of individual parts. However, it appears that the best power to weight efficiency will be obtained by having a minimum number of deployment mechanisms. For the purpose of this study, it has been specified that there will be two arrays each of 10 KW capacity in the system.

Interface with the spacecraft can have a significant effect on the structure and weight of system. Rigid mounting directly on the spacecraft structure will permit lighter weight structures in the array system at the interface. Arrays which can be oriented constantly to the sun line through 180° to 360° will require more structural strength within the stowage subsystem itself, and hence present a larger weight reduction problem.

2.3.1.2 Panel and Stowage Configuration

There are a number of panel and stowage configurations which merit consideration in this study. Among these types are rigid folding panels, semi-rigid hinged modules, flexible folding modules, and continuous blanket substrates. The method of stowage is directly tied in with the construction of the array and hence becomes a prime consideration in the design of a lightweight extendible array system.

Rigid folding panels are fabricated by mounting cells on a low weight structural panel such as honeycomb material with hinges on opposite sides. This method of construction is often adequate for small arrays. Although some good advances have been made, such as the EOS hollow core panel, rigid panels can be expected to be a limiting item in optimizing weight for large arrays.

Semi-rigid panels formed in modules and hinged for folding into a flat pack were selected for the SEPS array by LMSC.[1] This method was chosen because of its (1) module compatibility, (2) non-sensitivity to misalignment, (3) control and care of supporting cells in a flat condition for retention of the array under launch environments, (4) a flexible stowed volume, and (5) elimination of power transfer devices.

In optimizing weight during this study, tradeoffs will be made between the flat pack and rollup methods of stowage. The drum type rollup device is attractive because it provides a method of imparting continuous tension on the blanket for both extension and retraction by the addition of a simple negator spring motor to the drum. However, the drum approach has two limiting factors, namely (1) the tendency to bend thin cells assemblies about the radius of the drum, and (2) low weight and volume efficiency because of the wasted space inside the cylinder. The drum is efficient structurally, however, and this factor tends to overcome the other limitations.

2.3.2 LOW WEIGHT DESIGN APPROACH

In order to identify the most promising areas for weight reduction on this solar array system, particular attention has been placed on the weight distribution of current developed arrays and the 110 watt/kg concept developed by General Electric in 1973. The system is divided into four subsystems for the purpose of a weight distributions study, namely (1) Electrical (Solar Cells, Substrate, etc.), (2) Stowage and Support, (3) Deployment and Retraction Mechanism, and (4) Retention and Release Mechanism. By inspection of the weight distribution in typical flexible solar array systems (see Table 2-10),

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[1] Reference 27

Table 2-10 Typical Weight Distribution of Solar Array Systems

Subsystem	GE Rollup Array		LMSC SEPS Array		GE 110 W/kg Concept	
	kg	% of Total	kg	% of Total	kg	% of Total
1. Electrical	21.12	56.5	122.5	64	48.5	55
2. Stowage & Support	10.96	29.3	34.7	18	30.6	35
3. Deployment Mechanism	5.32	14.2	35.1	18	8.4	10
TOTAL	37.4	100	192.3	100	87.5	100

maximum reduction of weight should be obtainable by changes in the solar array blanket and the stowage and support apparatus. These are the highest weight percentage areas.

This study is made on the basis of an advanced silicon cell which has been defined by JPL as a projection of the state of the art in the 1977-78 time frame. This cell, resulting in a 40% reduction in cell thickness and a 36% increase in solar cell electrical conversion efficiency over the 110 W/kg array baseline design, results in an immediate weight savings of 48% in this predominantly heavy electrical subsystem. Since it follows that very light arrays will likely be very thin arrays, it appears that the advanced array will inherently be highly flexible.

2.3.3 PRELIMINARY DESIGN CONSIDERATIONS

With this trend toward a highly flexible blanket, some preliminary concepts are being explored. The following paragraphs discuss some of these concepts.

2.3.3.1 Flat Sided Rollup

If drum storage is employed, there will be a tendency to bend the thin cells assemblies to some radius which is a function of the cell stiffness, the blanket tension and the drum radius. Therefore the minimum radius which is acceptable in respect to bending of solar cell assemblies is of prime importance. The flat sided drum is a concept aimed at combining the features of the rollup approach with the desirable features of the flat pack; namely the non-bending of the cell assemblies. Having flats on the drum will cause a fluctuation in blanket tension during deployment and retraction if a constant tension torque is applied to the drum. The design curves in Figure 2-9 may be used for selecting an acceptable trade. It is desirable to keep the flat

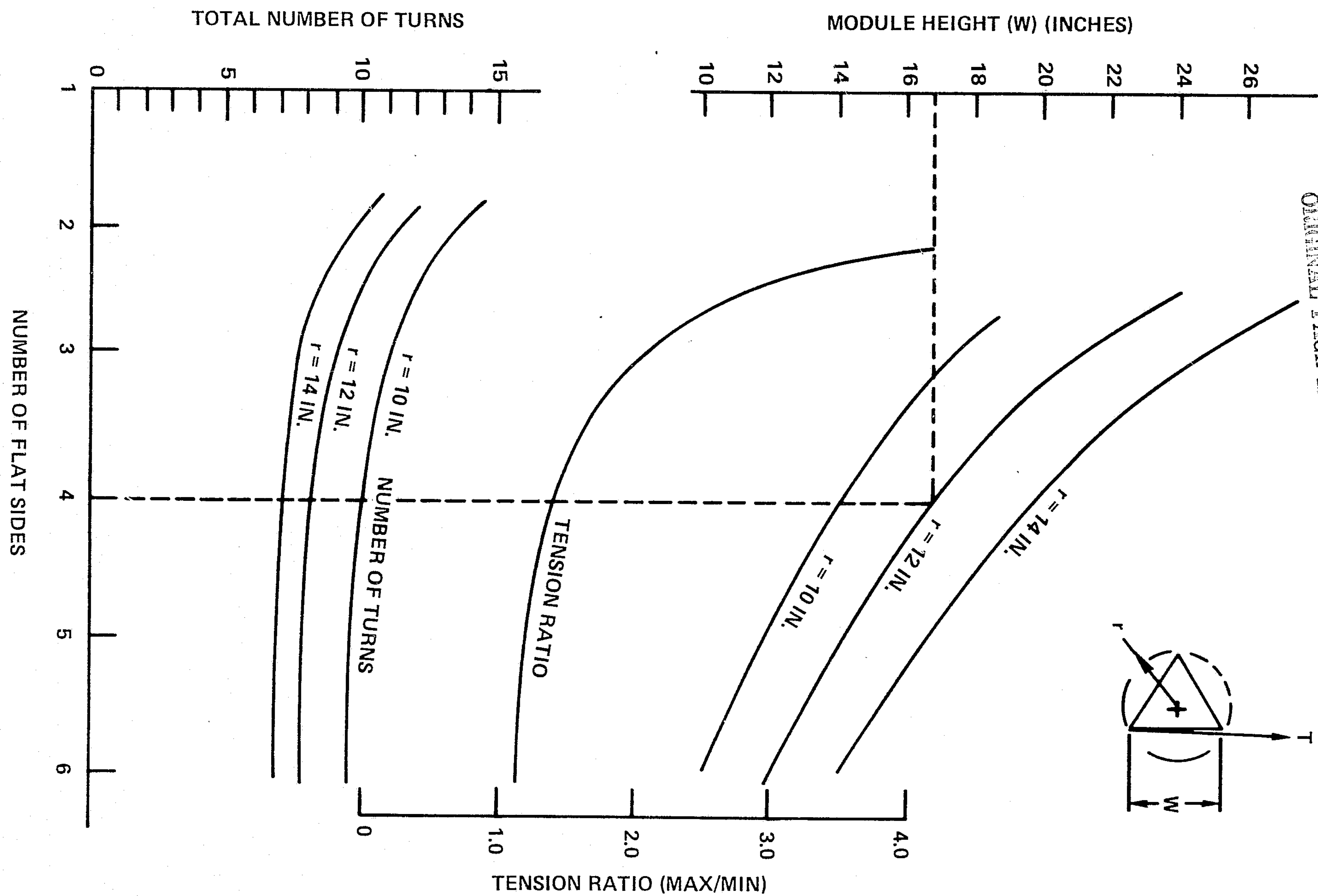

Figure 2-9 Trade-Off Chart for Flat Roll Up Arrays

width "W" large so that a reasonable module height on the blanket will be attained. But it is necessary to keep the number of flats on the drum high enough to keep the tension fluctuation down to an acceptable value. The dotted line represents a logical selection with 4 sides and a tension ratio of 1.4:1. This method of course requires that a small section of bare substrate exist at the corners of the rotor dividing the cell modules from each other. The total length of the blanket must be increased to allow for this loss in effective cell mounting area.

The minimum safe stowage bending radius for a 3 mil cell assembly is not easily determined at this time. Three mil cells and corresponding blanket assemblies do not currently exist. A minimum bending radius which will not result in excessive breakage of cells is dependent on finish conditions of the silicon cell, method of cutting, nature of cover material, adhesives, temperature conditions, etc. Further investigation of this important question will be conducted. Some data taken at GE Silicon Products Department, Syracuse, N.Y.[2] indicates that the fracture strain of 6 mil silicon ribbon is .0011 to .0014 in/in. The corresponding fracture stress values will be dependent upon the representative modulus of elasticity. In Figure 2-10, the theoretical strain and stress versus bend radius has been plotted for 3 mil thickness of thin sheet whose elastic modulus is 10×10^6 lb./in^2. It's for only a single layer sheet. When the solar cell assembly consists of a sandwich type construction, the outer fiber (point of max. strain) may exist in the cover glass and hence, maximum stress will probably occur at that point. The determination of minimum safe bend radius may be approached analytically but should be verified by testing the specific components, materials and construction involved in the cell assembly design.

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[2] Reference 3

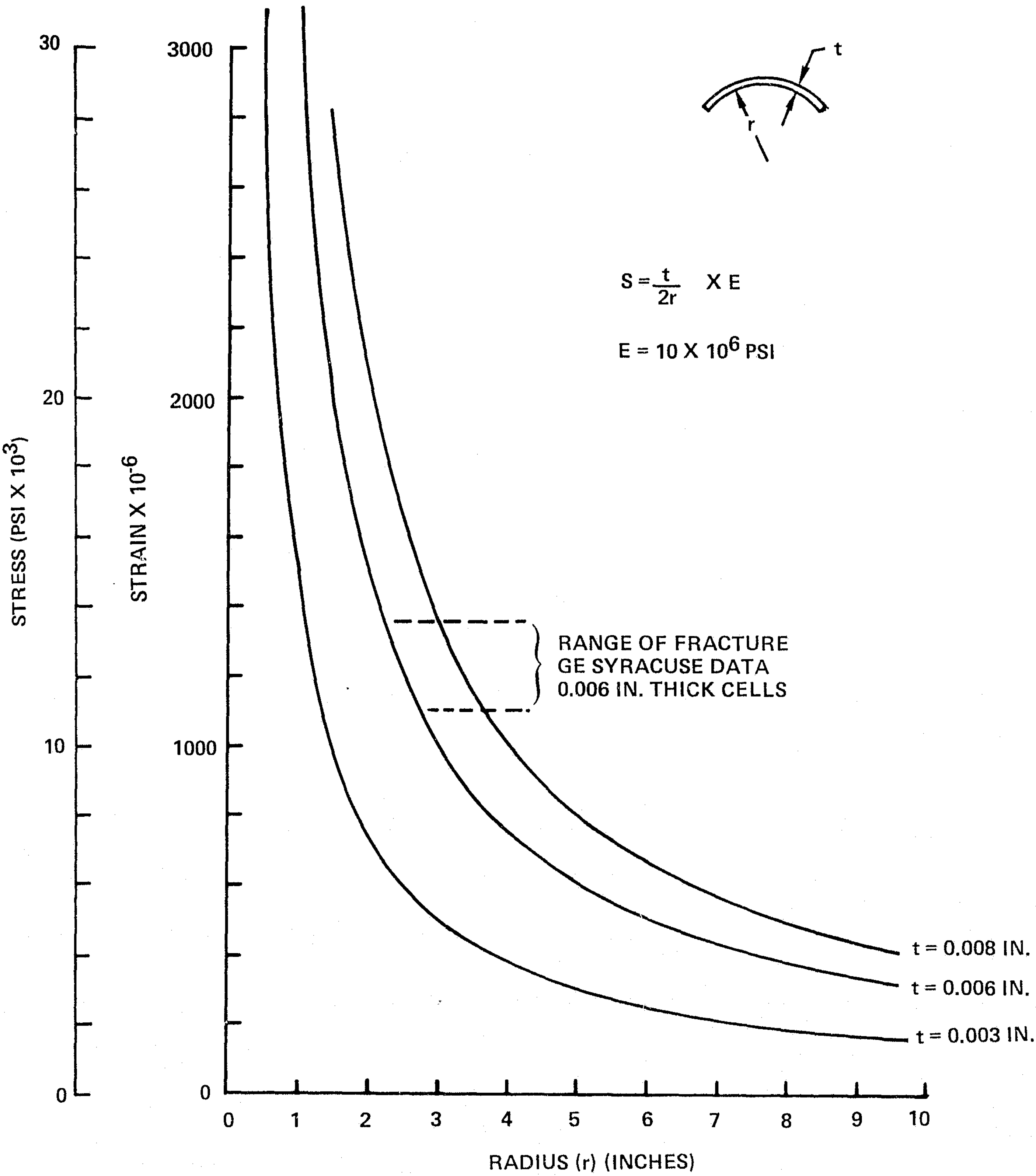


Figure 2-10 Stress and Strain In Bending of Thin Sheet Silicon

2.3.2.2 V-Stiffened Concept

Another approach in designing an array which will help in the task of weight optimization, is the V-stiffened concept first presented in the GE 110 W/kg study[3]. In this concept, the two blankets shown in Figure 2-1 of section 2.2.2 as being coplanar, are canted by a small angle so that tension forces in the blanket add to the bending stiffness of the overall mast array assembly (see Figure 2-11). Thus, a lighter weight mast is permissible. Preliminary structural analysis of a V-stiffened 200 W/kg array has been performed and is discussed in para. 2.5.3.

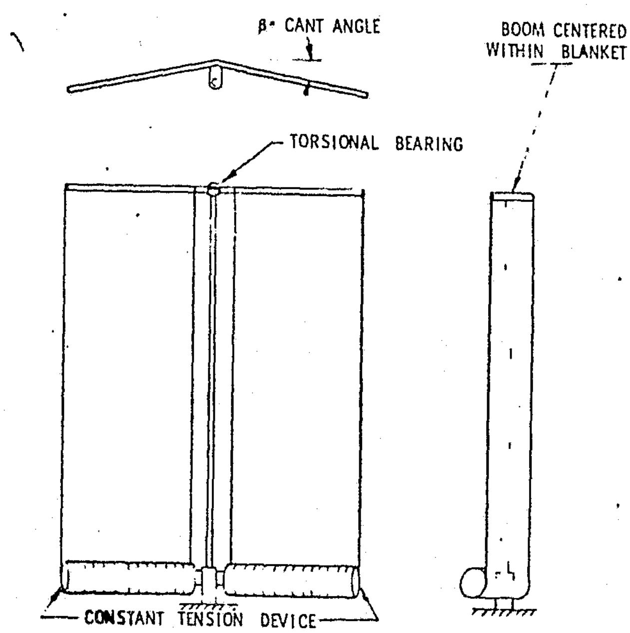


Figure 2-11 "V" Configuration, Single Boom Solar Array Concept

[3]Reference 1

2.4 EXISTING COMPONENT TECHNOLOGY BASE

2.4.1 GENERAL

A review of the existing technology base in the areas of solar cells, solar cell covers, interconnects and substrates, and deployable booms was performed during this reporting period in order to have an early assessment of the present state-of-the-art. This section discusses each of these areas with particular emphasis on the applicability to this study.

2.4.2 SOLAR CELLS

2.4.2.1 Introduction

The attainment of the specific power goal of 200 W/kg is dependent in large part on the reduction of the total weight of the solar cells required to output 10.2 KW, BOL (assumes 2% electrical loss). Cell efficiency, cell thickness, and configuration will all impact the requirement for light weight. Cell efficiency for any given cell configuration is a function of cell temperature and optical power density (insolation) incident on the cell. For the purposes of this conceptual design the study, the insolation will be held fixed at 1 A.U. (distance from the sun). This spectral power density for this insolation is defined by ASTM Standard Specification E490-73a. (Attached as Appendix B).

2.4.2.2 Cell Efficiency vs. Cell Thickness

Under the conditions of solar irradiance at 1 A.U. distance, the impact of cell thickness on cell efficiency is indicated in Figure 2-12. The fall-off in power output as the cell thickness decreases is associated with the low absorptivity of red and infrared photons in silicon. At the baseline thickness of 3 mils (75 μm), the cell power output has dropped to 89.5% that of a 10 mil cell. Conversely, 11.7% more cell area will be required to provide the same power

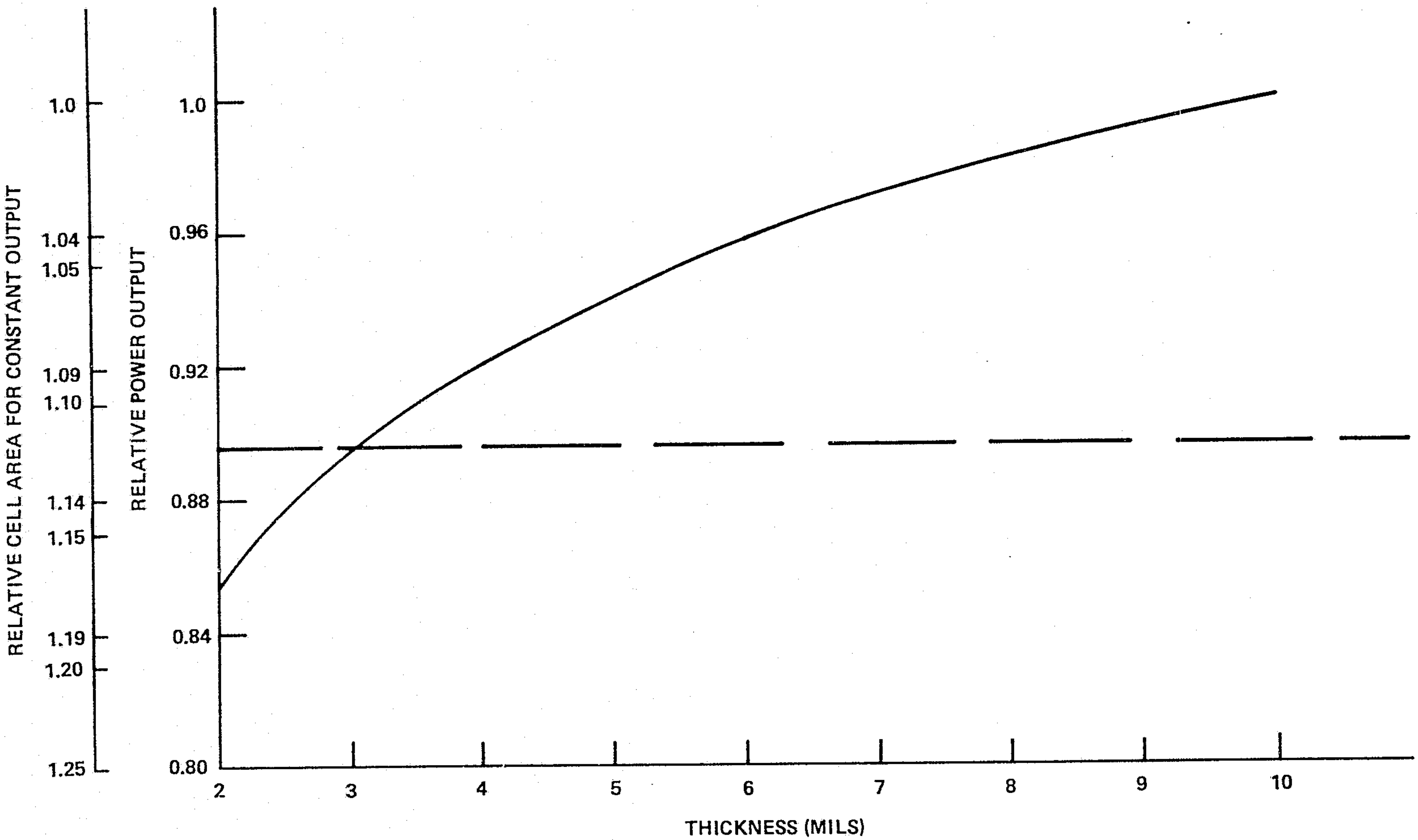


Figure 2-12 Cell Power vs. Cell Thickness

output as a 10 mil cell. It would appear that there would be a clear advantage to thin cells over thick cells on a specific power basis, since the incremental power loss is more than offset by the incremental gain in cell area. The specific power advantage at 3 mil would appear to be 0.896/.03 $\simeq$ 3 times that at 10 mil. However, the weight of additional cell interconnects, blanket size, support mechanisms, etc. will have to be accounted for before a true measure of the impact of the cell thickness factor on specific power may be made. An overriding consideration regarding cell thickness is that of breakage. It is a widely held observation that the thinner the cell the higher the breakage rate due to handling. In order to assess the potential of achieving an output power to weight ratio of 200 Watts/kg for a flexible roll-up array of Silicon, the glass cover slip which contacts the solar cell, and the composite mechanical nature of the system. This report will attempt to shed light on the scope of the problem and make conclusions concerning those elements which a rational approach, aimed at getting the necessary answers, might include. Aside from the unfortunate paucity of information on the mechanical properties of thin Silicon solar cell material; the other outstanding aspect of this topic is the extent to which disagreement exists, and the amount of variability in existing data. One reference shows a 50% variation in breaking strengths for single crystal Silicon of small dimensions (0.25 in. x 0.25 in. x 0.025 in.) with the same surface preparation. There is also no general agreement in the literature on just what value should be accepted for this material property. Source to source discrepancy may be due to differences in either the principal of measurement or the measurement technique. For instance, an early (1951) measurement of the elastic constant of Silicon was made acoustically. In this method, the elastic constant is calculated from the measured propogational velocity of sound in the material and the density of the material. More recently,

investigators have used the "three-point" loading technique where the sample is braced against two supports and loaded from beneath by a third moveable loading member. Results obtained in mechanical loading tests of this type will, to some extent, depend on the specifics of how the test was carried out and on the conditions at the boundaries where the sample under test is held to its support. There does, however, seem to be agreement that the failure property (fracture stress) will depart from its bulk value, and in some way be related to the surface quality of the material. It has been shown that Silicon wafers which have been chemically polished will withstand greater bending stress. The polishing eliminates many stress risers (i.e. minute cracks, notches and other surface imperfections) by simply etching below them. Silicon solar cells for lightweight space array applications are a composite of several different materials. It is really the mechanical (flexural) behavior of this composite that is of interest in this study. Pertinent information on this subject is for the most part not available. Therefore, such information must be empirically generated. In addition to the flexural properties of specific solar array materials and their composite behavior, attention should also be given to a consideration of "stress-time-temperature" effects. The design of all such studies should duplicate real loading and other ambient conditions as closely as possible in order that results obtained from such studies will predict the performance of the entire system under actual space flight conditions with minimum uncertainity.

TABLE 2-11

TEMPERATURE DEPENDENCY

OF

CELL PARAMETERS

VOLTAGE:	$\Delta V/\Delta T$	=	$-2M\ V/^{o}C$
CURRENT:	$\Delta I/\Delta T$	=	$30\mu\ A/Cm^{2}-{}^{o}C$
POWER:	$\Delta P/\Delta T$	=	$270\mu\ W/^{o}C$*
	ARRAY PARAMETERS		
VOLTAGE:	$\Delta V/Vo$	=	$-0.4\%\Delta T$
	$\Delta P/P$	=	$-0.33\%\Delta T$

* 2 x 2 cm^2 area cell

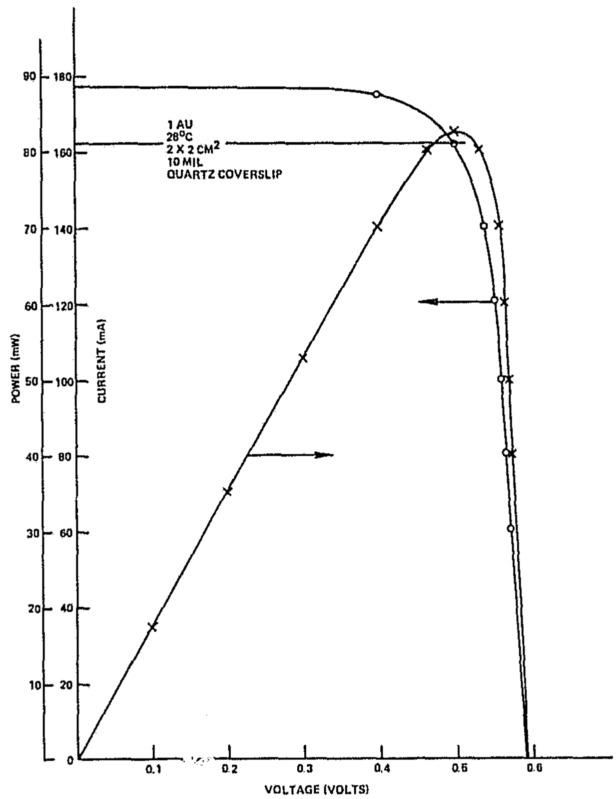


Figure 2-13. Candidate Development Cell Data

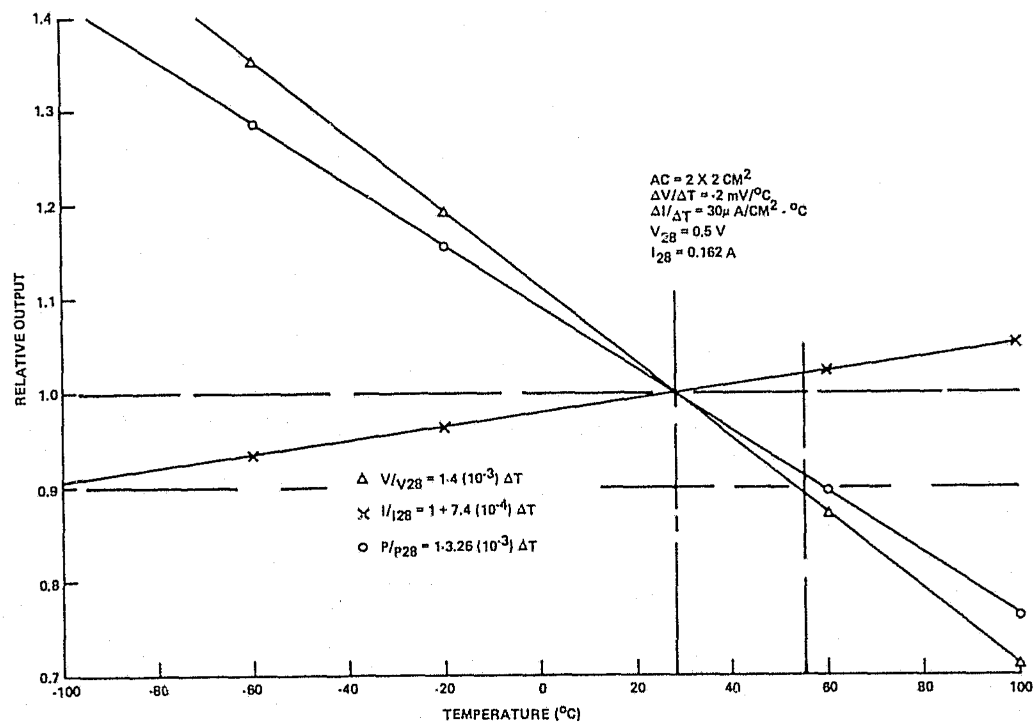


Figure 2-14. Temperature Dependency of Cell Characteristics

2.4.2.3 Cell Temperature Dependence

For the purposes of this study, a current/voltage performance characteristic has been assumed for a 2x2 cm cell, 10 mils thick. This characteristic and the associated power output curve is shown in Figure 2-12 for the standard condition of insolation and temperature. Additional assumptions have been made regarding the temperature dependence of voltage, current and power, see Table 2-11. Using the candidate cell data of Figure 2-13 and these temperature coefficients, a prediction of cell performance as a function of temperature is shown in Figure 2-14.

2.4.2.4 Radiation Hardening

The Baseline Requirement document states that the solar cell must successfully withstand a total dosage of 2 $x(10^{12})$ proton (1 MEV) particles per cm^2 over 3 years. The radiation damage to be expected at this dosage will be the subject of analysis in the next quarter of the program.

2.4.2.5 Configuration

A configuration trade off analysis will be conducted during the next quarter of the program.

2.4.3 CELL COVERGLASSES

2.4.3.1 Introduction

An efficient coverglass serves several purposes; namely, as a thick barrier to physical particles, a radiation shield for high energy particles, a UV baffle, low index of refraction optical interface to air/vacuum, and a good radiator for thermal energy. The density of the most commonly used coverglass material (fused silica) is almost on a par with silicon (2.2 g/cm^3 vs. 2.3 g/cm^3 for silicon). A thermoplastic material, FEP (Fluorinated ethylene propolene), is often cited as having suitable properties.

Its density is 2.15 g/cm^3, about 2% lighter than fused silica. The battle of weight reduction at the coverglass interface will be won by finding a substitute material with suitable physical and thermal properties or by application of very thin sections of the standard materials.

2.4.3.2 Material Investigation

A search for appropriate material and methods of applying these materials will be made in the next quarter of this conceptual design study. A leading candidate at this time would appear to be ion sputtered or RF sputtered fused silica. A second contender would appear to be 1 to 2 mil FEP. In either case, the principal criteria to be used in candidate evaluation include resistant to 1) UV radiation, (2.0 calories/cm^2-min for 1095 days), 2) radiation dosage of 10^7 RADS and 3) high thermal emissivity at cell operating temperature. The effectiveness of the material as a barrier to high energy particles shall not be at issue, since solar flare proton fluency has been disregarded as a source of solar cell power loss. Refer to paragraph 3.2.3 of Baseline Requirements, Appendix A.

2.4.4 CELL INTERCONNECT AND SUBSTRATES

2.4.4.1 Introduction

The conceptual design of low resistance, highly reliable cell interconnects and substrates will be based on the accumulated past experience of many space power systems, influenced by the present specific power requirement of 200 W/kg. The integrity of the ohmic contact during thermal cycling is of particular concern. The selection of substrate material will be made on the basis of resistance to particulate surface dosage radiation of 10^7 rads and UV exposure equal to 1095 days at the rate of 2 cal/cm^2-min.

2.4.4.2 Material Investigation

A review of materials previously identified by studies and/or usage will be made

in the light of the present requirement on specific power. The integrity of physical properties over a 3 year mission, good thermal emissivity, and a high strength to weight ratio are the principal criteria.

2.4.5 DEPLOYABLE BOOMS

Extensive surveys of deployable booms have been made in the past, particularly in connection with the Space Station Array Program, and are well documented. This study will utilize these findings and review any advancements of the state-of-the art of extendible members in respect to their significance to the current study. The deployable booms, which appear to be good candidates at present, are (1) the Astromast Coilable Lattice Boom, Figure 2-11, made by Astro Research Corporation, and (2) the BI-STEM (Storable Tubular Extendible Member) boom, Figures 2-15 and 2-1, also manufactured by Astro Research. The Coilable Lattice Boom has continuous longeron members running the entire length of the mast. When retracted, these longerons coil on top of each other as shown in Figure 2-16. When longeron stresses generated by the coiling of the boom are excessive, as in case of high temperature, segmented steel longeron members are utilized. These fold on retraction. This version is the articulated lattice structure shown in Figure 2-17. In general, lattice structures have the advantage of a high strength to weight ratio in applications where a relatively high EI (modules of elasticity) and bending strength are needed. Their disadvantage is the large stowage volume and actuator weight required. The widely used BI-STEM boom is essentially a ribbon of thin metallic material which assumes a tubular shape of high strength when unwound from a spool. Additional strength is obtained by using two tapes which form around each other as shown in Figure 2-18. The actuator has a motor driven spool and tubular guide mechanism. The unit shown in Figure 2-19 is the A-631 model containing a 1.34 inch diameter mast. This device is compact and weighs less than 6 pounds (less BI-STEM element).

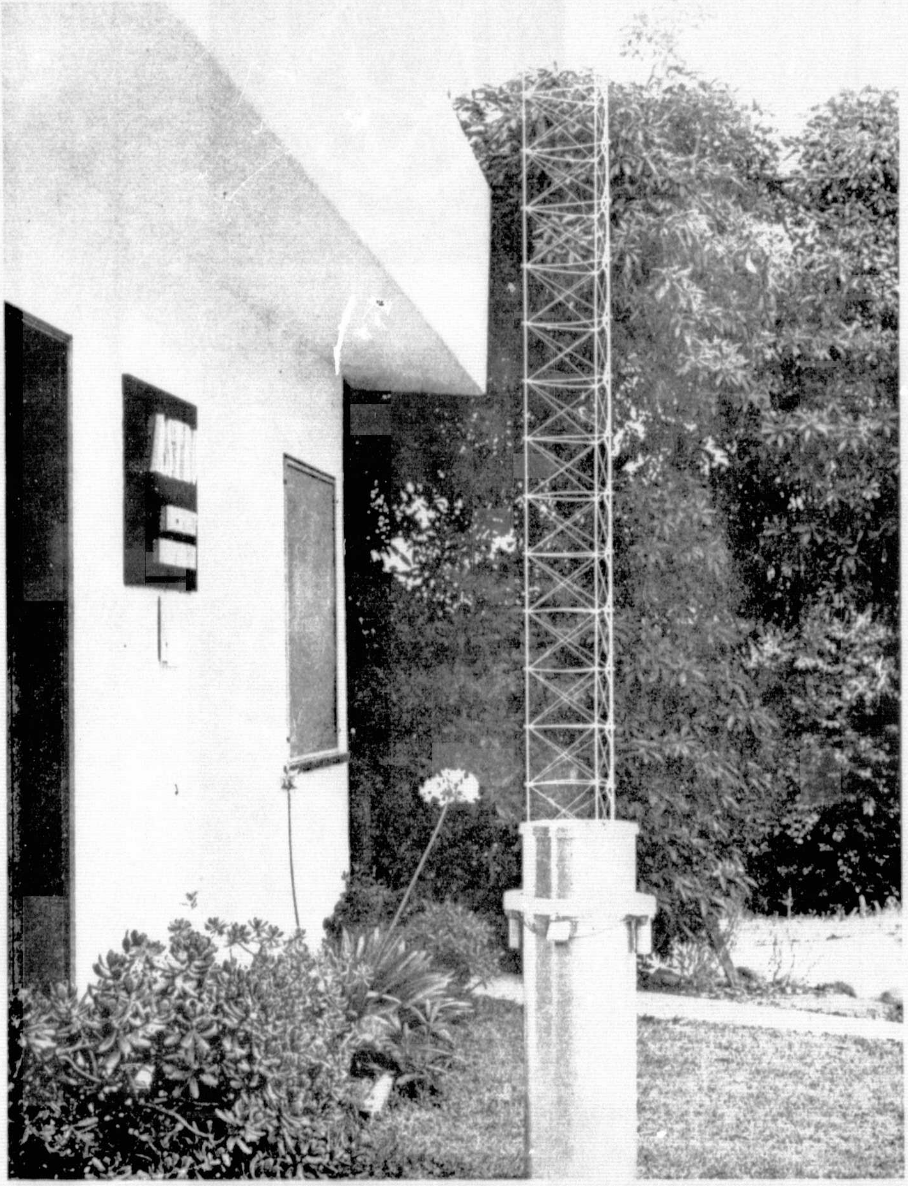
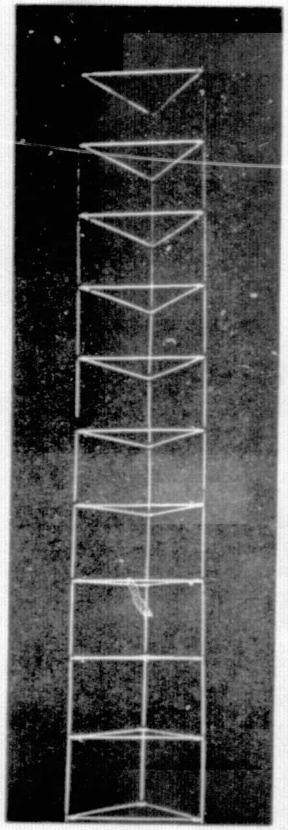
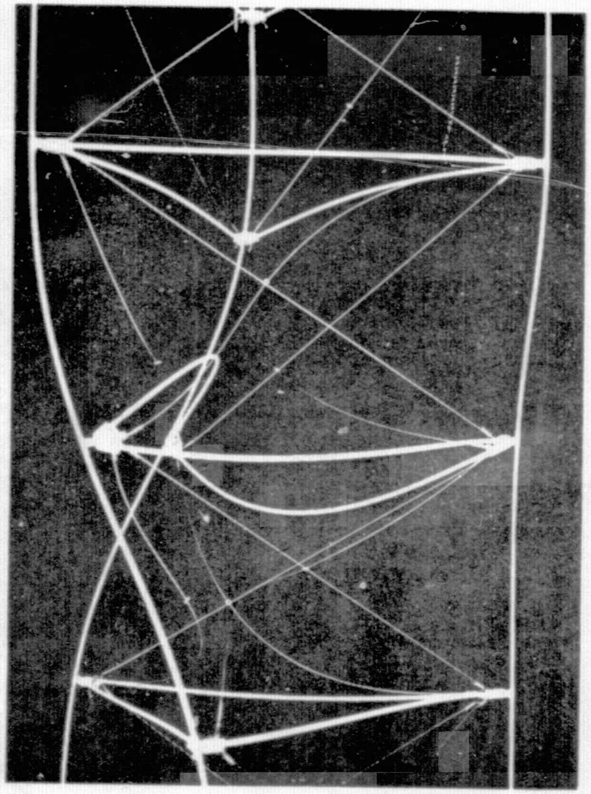


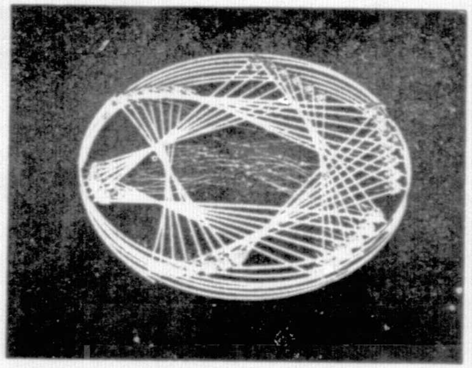
Figure 2-15. ASTROMAST Coilable Lattice Boom - Lunar Antenna Mast



Segment of Flexible
ASTROMAST Column

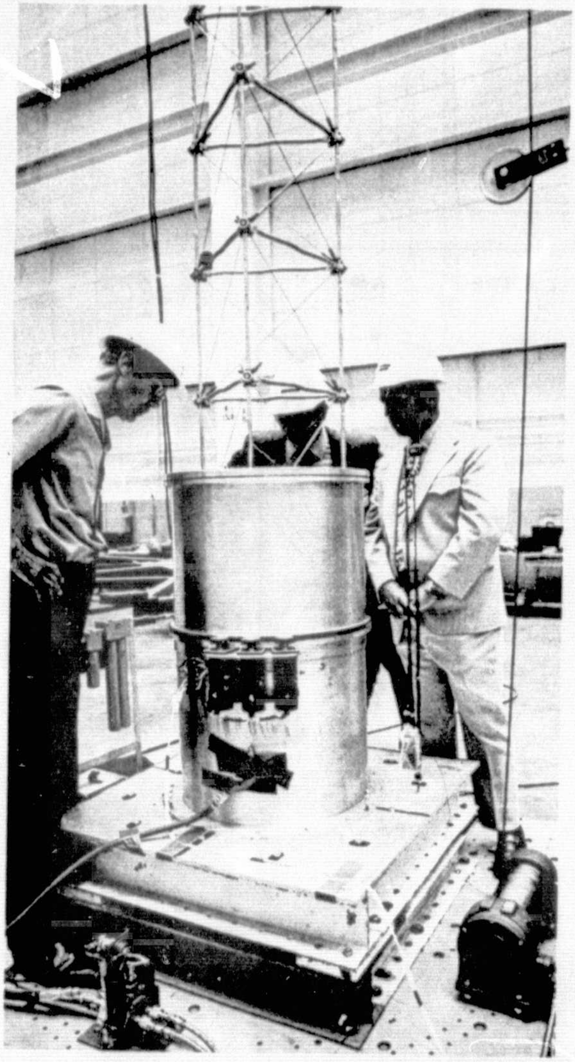


Buckled Battens of Column

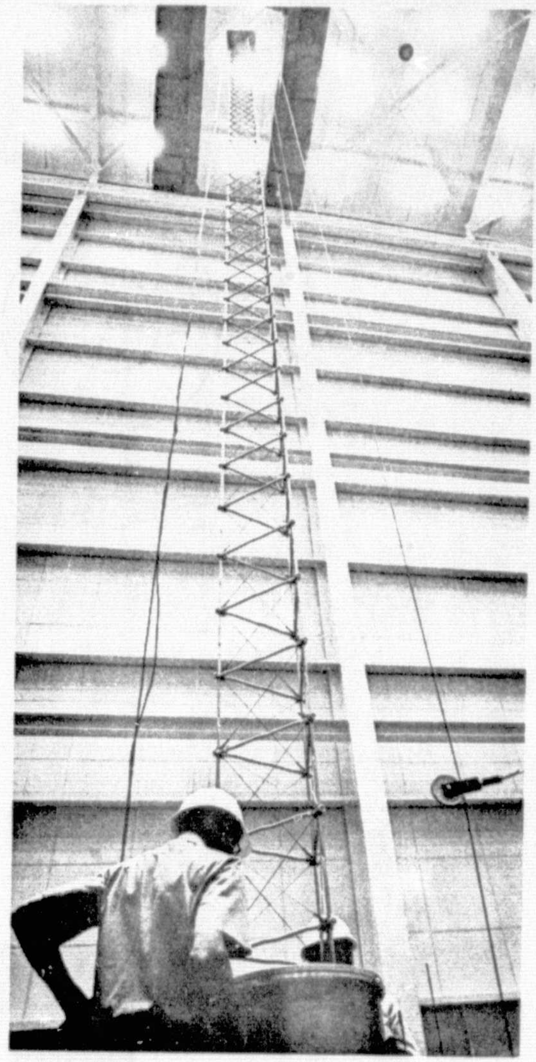


Packaged Configuration

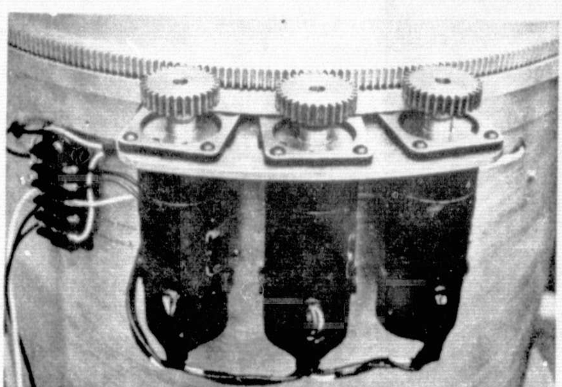
Figure 2-16. Continuous Longeron ASTROMAST Concept



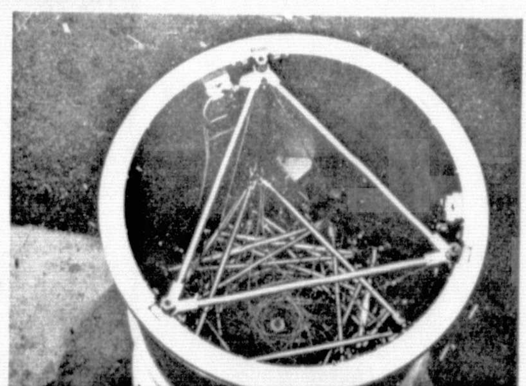
Astromast Deploying
Automatically



Fully Extended Mast



Deployment Motors



Retracted Configuration

Figure 2-17. ASTROMAST Articulated Lattice Boom
for Lockheed Space Station Solar Array

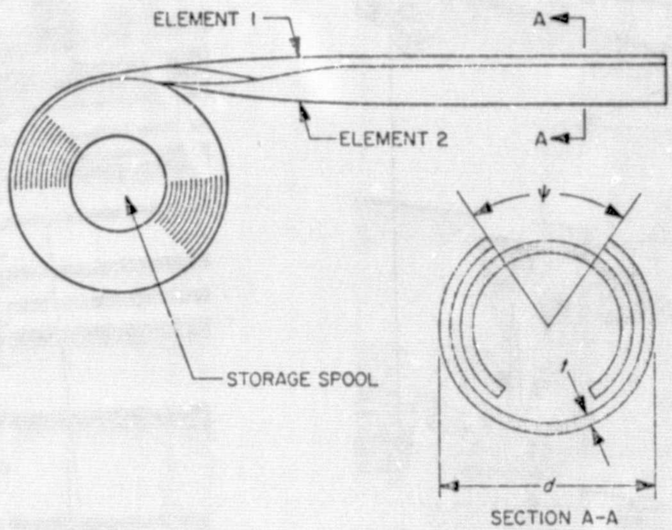


Figure 2-18. The BI-STEM Principle

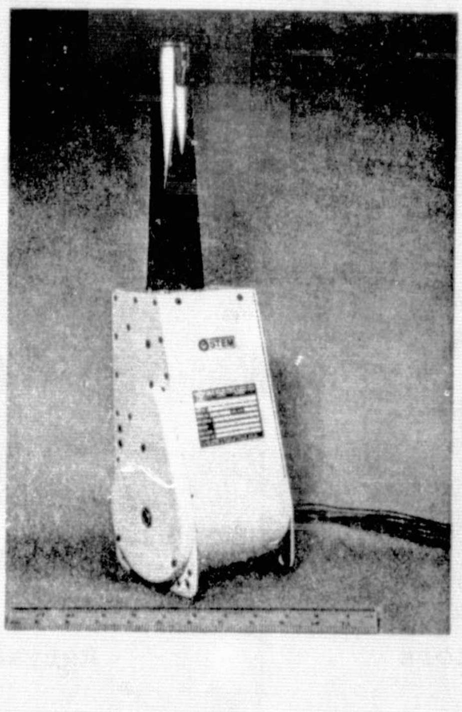


Figure 2-19. BI-STEM Deployable Boom and Actuator

2.5 PARAMETRIC ANALYSES

2.5.1 GENERAL

This section describes the dynamic parametric analysis performed on both a planar and V-stiffened array configurations. The results of the analyses and their impact on the design of a 200 watt per kilogram solar array are discussed.

2.5.2 PLANAR ARRAY - VIBRATION STUDIES

2.5.2.1 Introduction

Parametric studies performed during the 110 W/kg solar array study formed the basis for establishing the optimum structural configuration characteristics of the 200 W/kg array. The design requirements for an interplanetary mission were essentially the same with a minimum deployed array natural frequency of .04 Hertz and a deployed array quasi-static load capability of 1×10^{-3} g's. The launch environment is that of the Space Shuttle instead of the Titan-Centaur, but this change only affects the stowed configuration and is not extensively different. In this report, only the deployed array vibration studies are being considered.

The following discussion reviews much of the investigations, techniques, and optimization studies which were conducted on the 110 W/kg array. A preliminary configuration was selected based on the findings of the earlier study and the new guidelines for lighter weight, more efficient solar cells.

Computer codes for determining minimum boom bending stiffness and blanket tension to meet the .04 Hertz frequency were updated and results are presented for the new baseline configuration.

2.5.2.2 Background Data

The configuration selected for the 110 W/kg interplanetary mission is shown in Figure 2-1. It consists of two Kapton solar cell blankets supported by a tubular Beryllium leading edge member, and a single deployable articulated steel longeron ASTROMAST. A flat pack design was used for the stowed solar array. Table 2-12 presents a total system mass summary for the system. The total weight of 87.5 kilograms was split almost equally between the mass of the solar cell blankets and the supporting and packaging structure. The goal of 200 W/kg necessitates the 87.5 kilograms be reduced to about 50 kilograms and indicates the need to reduce weight in all areas.

Figures 2-20 and 2-21 present the first anti-symmetric (torsion) and symmetric (bending) frequency versus blanket tension characteristics for a planar array. As indicated by the curves, their crossover point gives the maximum natural frequency for a minimum blanket tension. Figures 2-22, 2-23, and 2-24 graphically present the results of optimized boom stiffness and tension versus natural frequency and minimum system weight. The overall conclusion derived from these parametric variation studies was that a three to one aspect ratio was the minimum weight design for baseline configuration shown in Figure 2-1.

In view of the extensive optimization analyses performed during the 110 W/kg study, a planar baseline configuration was selected which closely matched the findings from the earlier investigation. However, it is shown in subsequent discussions that the optimized planar array with just the lighter and more efficient solar cells does not meet the desired weight goal of 50 kilograms. Further weight reductions and/or configuration changes are indicated.

Table 2-12 Total System Mass Summary - 110 W/kg Array
(Baseline Configuration for Interplanetary Mission)

Item		Mass (kg)
Solar Cell Blankets (see Table 2-6 for detail breakdown)		48.5
Stowage and Support Structure		30.6
Frame	11.0	
Container Bottom	3.6	
Container Cover	4.0	
Container Mechanisms	0.1	
Center Fitting	0.8	
Leading Edge Member	3.1	
End Retention Fittings	1.0	
End Retention Cable Cutters	0.9	
End Retention Mechanisms	0.4	
Blanket Tension Mechanisms	1.2	
Interlayer Cushioning	2.5	
Container Foam	1.8	
Coatings	0.1	
Fasteners	0.1	
Deployment Mechanism		8.4
Mast	3.1	
Canister	5.3	
	Total	87.5

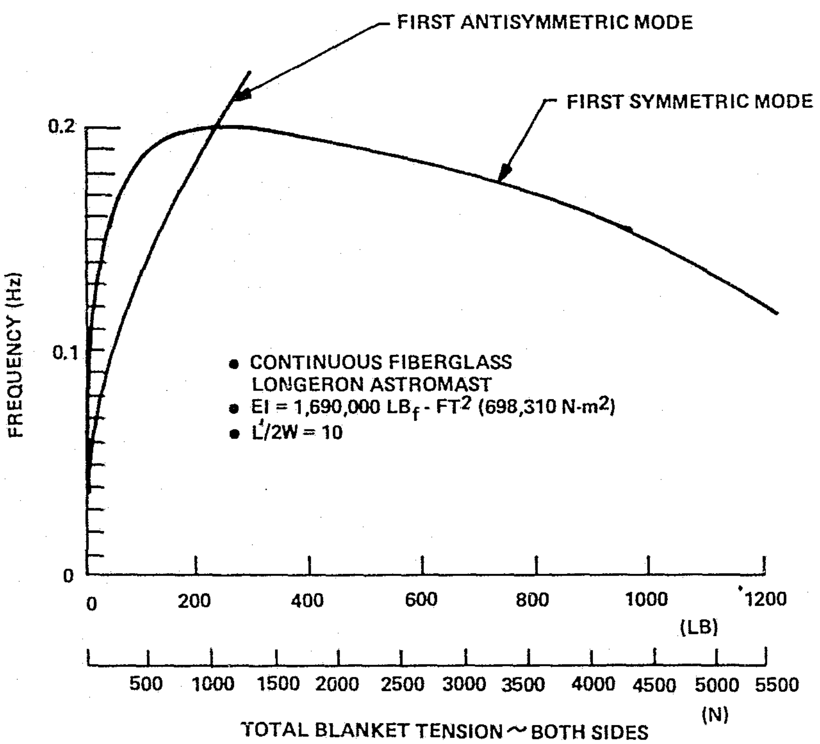


Figure 2-20 Symmetric and Antisymmetric Frequency Vs. Tension (ASTROMAST)

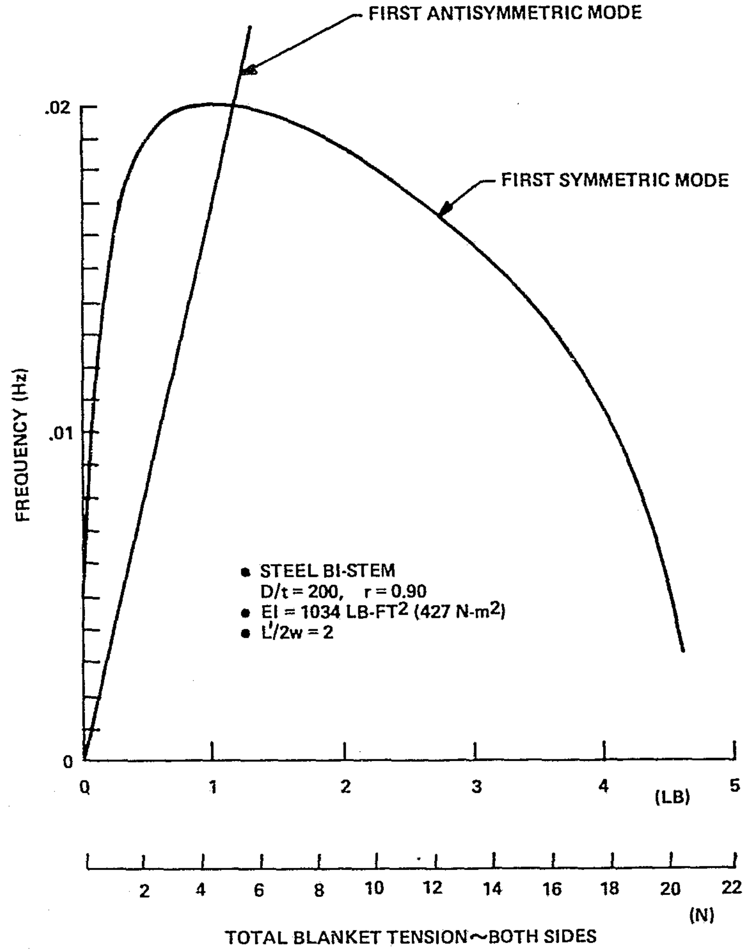


Figure 2-21 Symmetric and Antisymmetric Frequency Vs. Tension (BI-STEM)

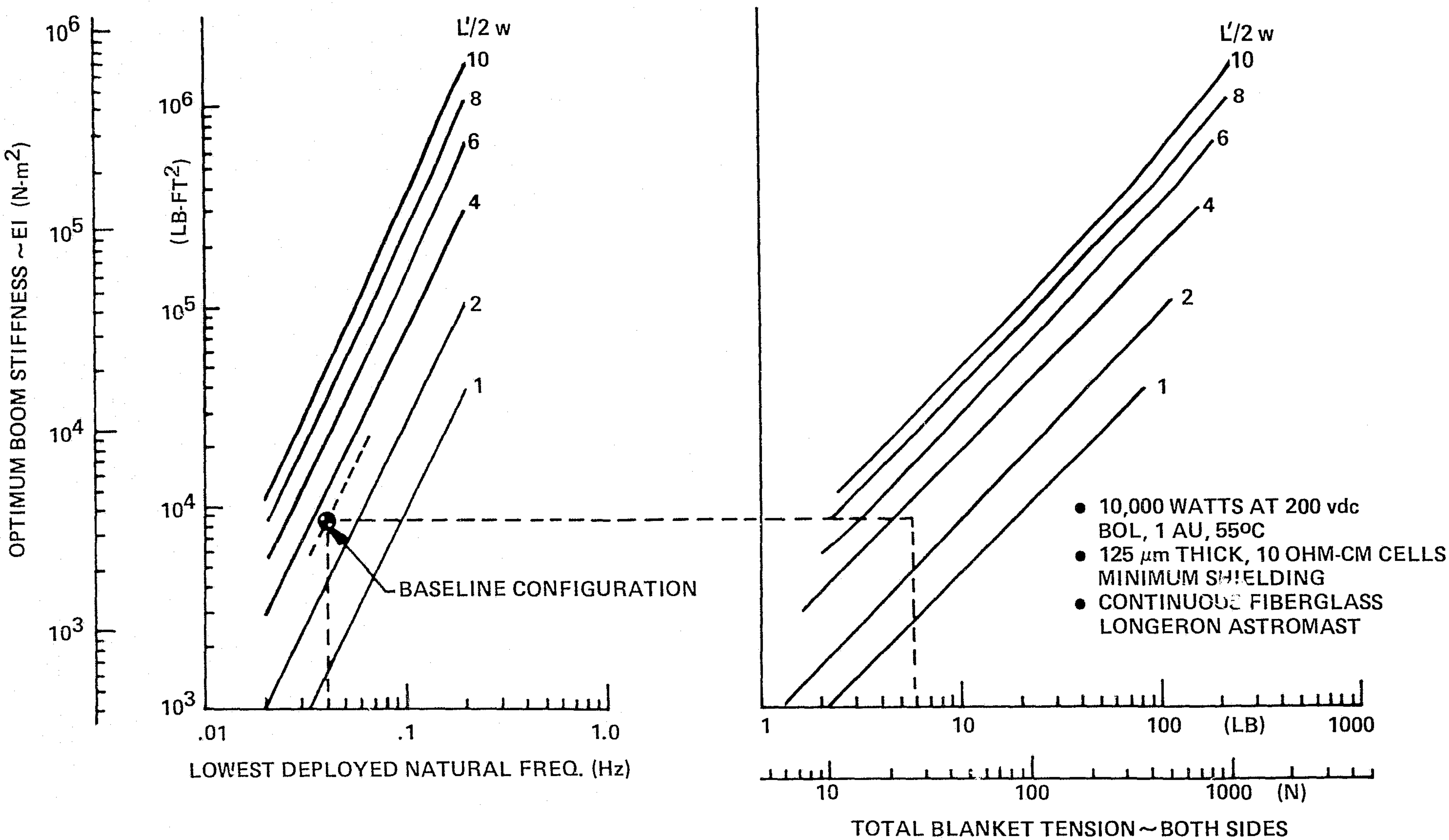


Figure 2-22 Optimum Boom Stiffness and Tension Vs. Frequency

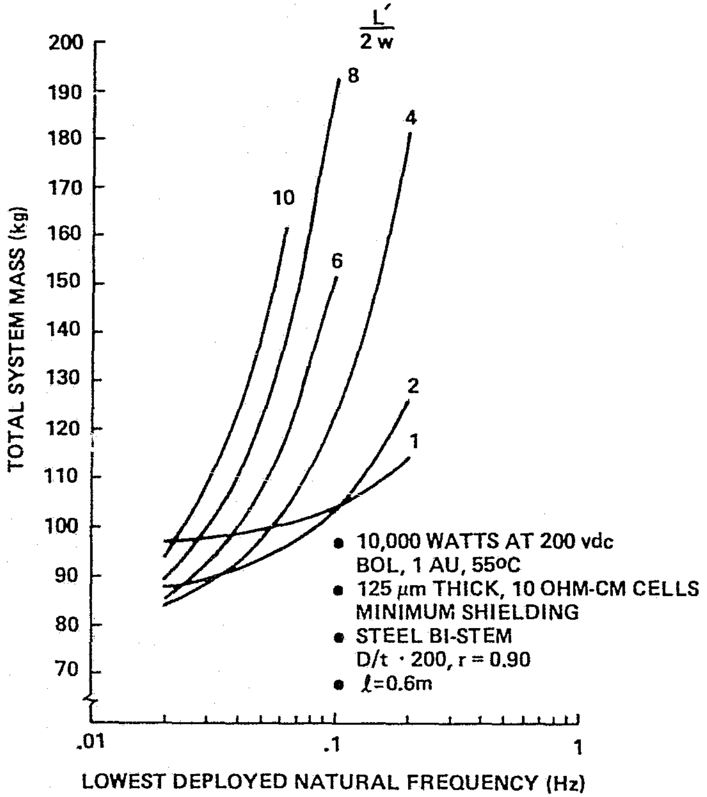


Figure 2-23 Total System Mass Vs. Deployed Natural Frequency ASTROMAST

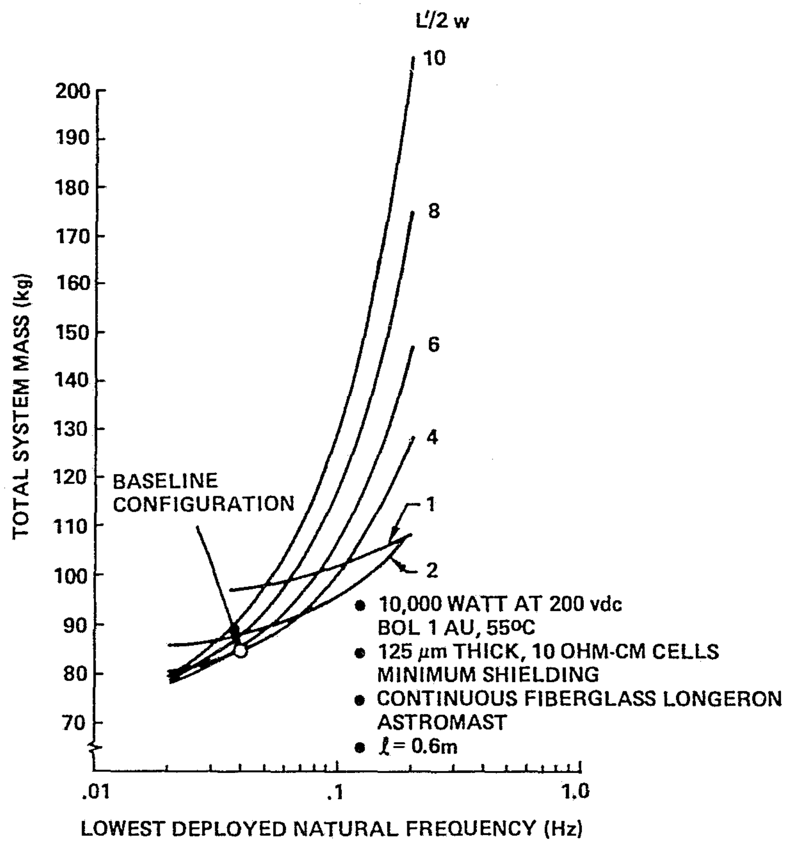


Figure 2-24 Total System Mass Vs. Deployed Natural Frequency BI-STEM

2.5.2.3 Planar Array Computer Codes

The dynamics analysis required to determine the optimum boom stiffness and tension of the planar array was performed using a discrete parameter model used for previous analyses on the 30 W/lb. and 110 W/kg studies, and verified by test. The model used a five by two discretization as shown in Figure 2-25. Because of the symmetry of the solar array configuration, only half the array was analyzed with appropriate boundary conditions to determine either the symmetric or antisymmetric array modes. Each blanket was represented by 10 rectangular elements that describe the out-of-plane stiffness caused by the blanket tension. The leading edge member (LEM) and boom were modeled using beam elements and included the effect of axial preload on the boom stiffness. The leading edge member was free to rotate relative to the boom about the longitudinal axis of the array. A consistent mass representation was used. The boom density was varied in accordance with the boom stiffness as shown in Figure 2-20 for the continuous longeron ASTROMAST and articulated steel ASTROMAST booms. The analyses were performed using the appropriate subroutines in a DYNAMO II program that enabled the parameters to be varied over the range of interest. (See Table 2-13)

2.5.2.4 Deployed Analysis

2.5.2.4.1 Optimum Blanket Tension

The initial analysis performed was to determine the optimum blanket tension necessary to meet the .04 Hz frequency requirements for both symmetric and torsional vibration. Since the LEM rotates freely on the end of the boom, boom stiffness does not affect the torsional frequency calculations of the array, and the tension required to meet the torsional frequency criteria can,

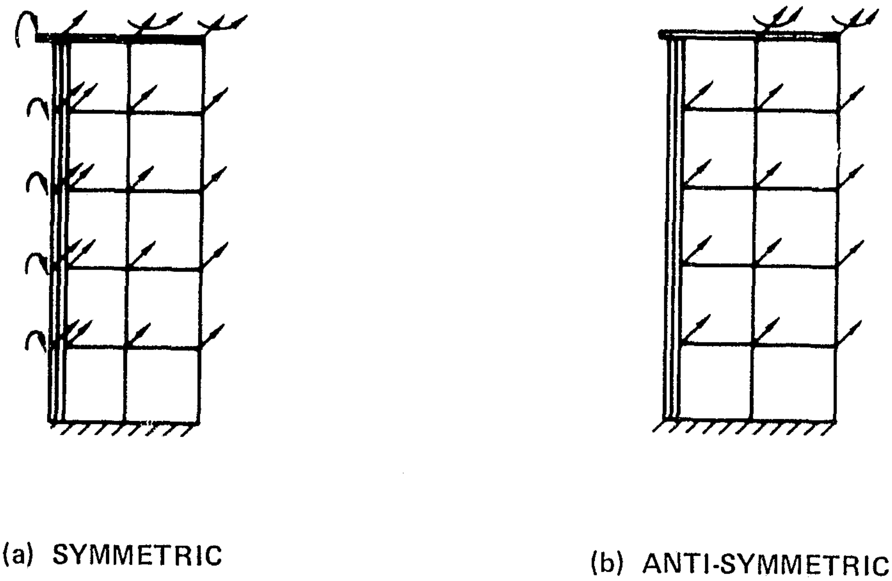


Figure 2-25 Finite-Element Model of Two-Blanket, Single-Boom Solar Array

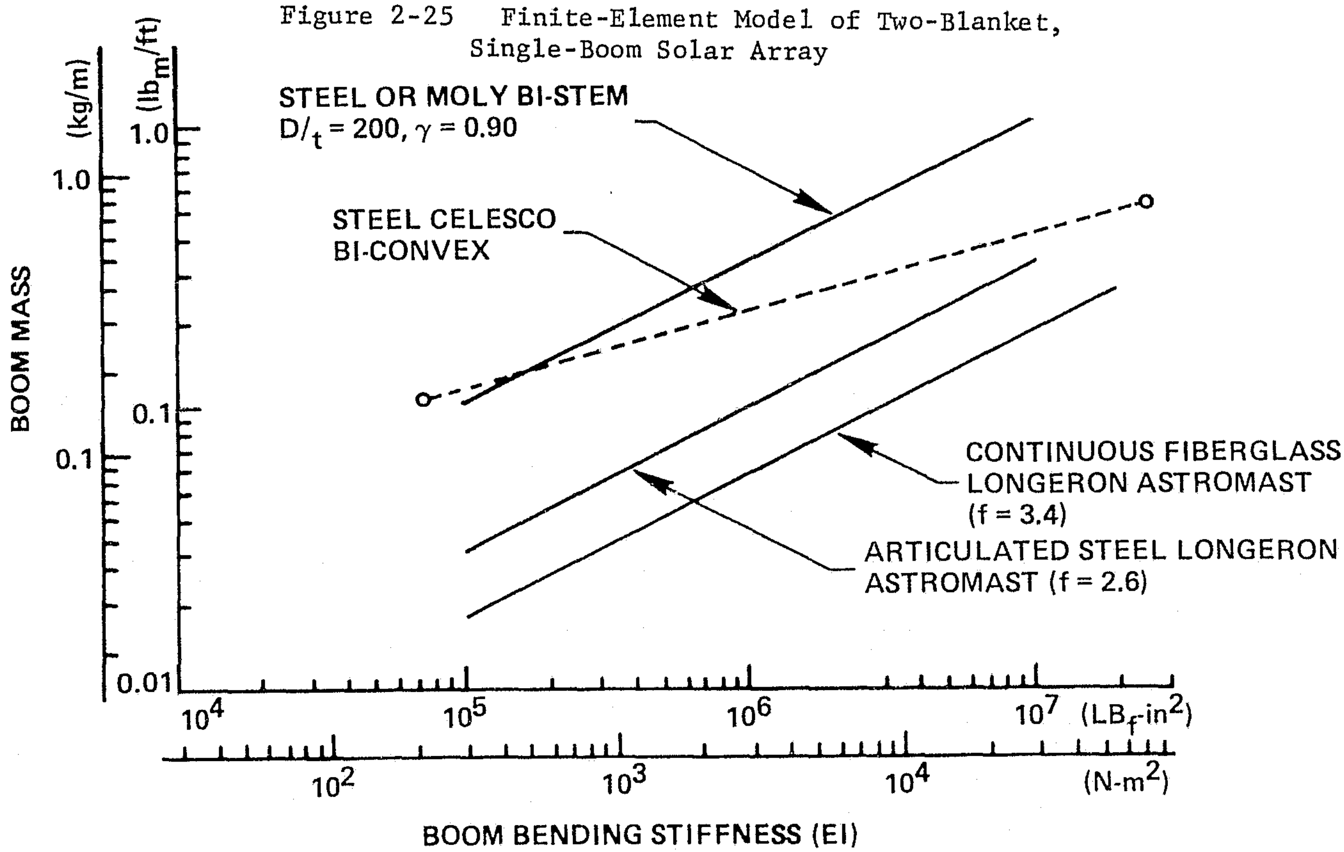


Figure 2-26 Deployable Boom Mass Vs. Bending Stiffness

therefore be determined solely from the information given above. This was done utilizing the rectangular membrane finite element program developed for determining the anti-symmetric frequencies of the 110 W/kg array. Once this value of tension was determined it was input, along with the data in Table 2-14, into the symmetric analysis program, to determine the boom stiffness necessary to meet the .04 Hz criteria for out-of-plane bending.

The results have been plotted in Figure 2-27 from which it can be seen that the optimum tension for the baseline system is approximately 2.4 lbs. (both sides).

Table 2-13 contains a summary of information acquired from the computer codes for use in the present 200 W/kg study.

Table 2-13 200 W/kg Baseline Planar Array Computer Runs

Run No.	
SAS002	Effect of tension on torsional frequencies.
SAS003	Effect of EI variation on symmetric frequenceis assuming: - Fiberglass ASTROMAST - Tension equal to that required for a torsional frequency of .04 Hz.
SAS004	Effect of EI variation on symmetric frequencies assuming: - Articulated steel ASTROMAST - Tension equal to that required for a torsional frequency of .04 Hz.
SAS005	Effect of tension variation on symmetric frequency.

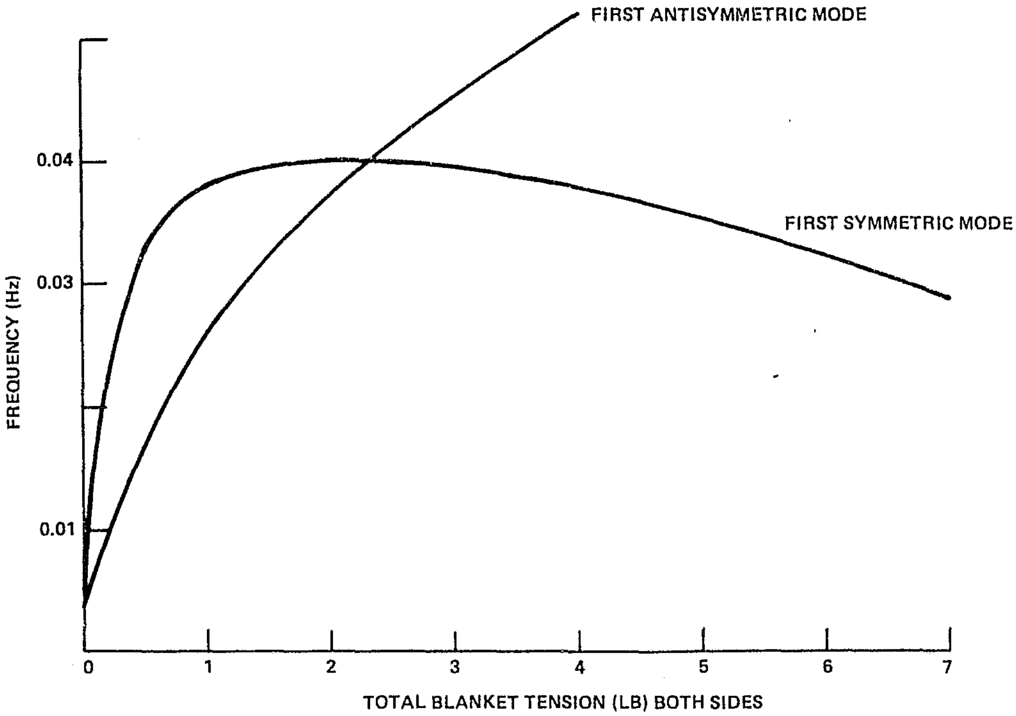


Figure 2-27 Effect of Blanket Tension On Lowest Natural Frequency

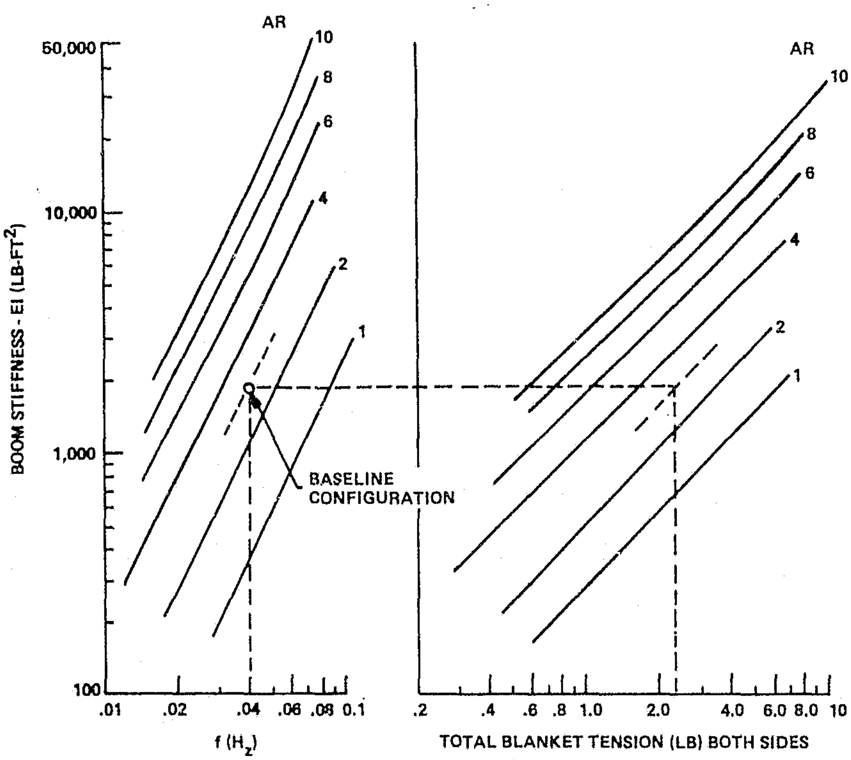


Figure 2-28 Boom Stiffness As A Function of Frequency and Tension

Table 2-14 Baseline Configuration - Planar Array

Aspect Radio	(AR)	3	
Length	(L)	14m	46 ft.
Width (total)	(W)	4.68m	15.3 ft.
Blanket Mass	(M)	24.97 kg	$55.06/g_c$ slugs
Blanket Density	(ρ)	.381 kg/m^2	$.078/g_c$ slugs
Required Frequency	(f)	.04 Hz	

2.5.2.4.2 Effect of Aspect Ratio

In this analysis, data calculated for the 110 W/kg study was modified to represent the 200 W/kg array. Figure 3-100 of the 110 W/kg final report[1] presents a plot of optimum boom stiffness (EI) vs. fundamental frequency (f) and optimum blanket tension for various aspect ratios. Since the primary equation in the determination of the tension required for .04 Hz in torsion is that for tension in a stretched string,

$$TS = \frac{4}{\pi}\frac{W^2}{2} X \delta L^2$$

the tension scale on this plot was shifted by ratioing the tension values by ρL^2 of the baseline 200 W/kg blanket to the ρL^2 of the 110 W/kg blanket. The boom stiffness scale of the plot was then similarly shifted by ratioing the values on this scale by TL^2. The results are shown in Figure 2-28 of this report.

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[1]Reference 1

Computer runs were made for the 200 W/kg baseline that verified that the results are sufficiently accurate for evaluation of the 200 watt/kg configurations.

2.5.3 V-STIFFENED ARRAY-VIBRATION STUDIES

2.5.3.1 Introduction

Parametric studies performed for the 110 W/kg solar array as well as the current 200 W/kg study indicated that further reductions in required boom stiffness and overall system weight can be obtained from a minor variation of the planar design to a canted or V-stiffened one. The 110 W/kg study was able to meet its system weight goal with a planar array. However, as shown in the previous section, the planar configuration cannot meet the system weight requirement unless a significant mass reduction is made in the stowage and support structure. The following discussion presents a review of the earlier investigations into V-stiffening effects and a 200 W/kg baseline configuration is established which nearly meets the system weight goal.

2.5.3.2 Background Data

A "V"-stiffened solar array configuration was conceived as a means of obtaining significant increases in the minimum array resonant frequency without added complexity. Thus, it is possible to meet a specified deployed natural frequency requirement with reduced boom stiffness (and reduced total system weight) when compared with a planar array geometry. This concept, shown in Figure 2-11, uses the slight angle of the array blankets to enable observed in-plane stiffening resulting from the redistribution of blanket tension to provide out-of-plane stiffness. Static tests and analysis of the in-plane behavior showed that the array blanket tension was redistributed such that the

array rotated about one edge. In effect, the blanket provided a moment constraint to the tip of the deployable boom until an edge tension condition was achieved after which the boom behaved as a cantilever. By canting the blankets and centering the boom within the blankets, this boom tip constraint can be used to stiffen the array for symmetric out-of-plane motion.

The effect of the canted blankets will also provide stiffening for torsional motion of the array. For a given boom, the tip constraint will enable greater tension to be applied without buckling the boom; hence, an increase in the torsional frequency. In addition, the boom will be required to bend during torsional vibration with some increase in the frequency. (Because of the high inplane stiffness, the array will tend to twist about the center of the "V" causing bending of the boom.)

Figure 2-29 graphically illustrates the effect of blanket tension on natural frequency for a "V"-stiffened array. There is no crossover or hump maximum frequency for the symmetric mode as exists for the planar array. Conversely, for a given frequency, "V"-stiffening permits the use of lower blanket tensions and subsequently lighter weight support structure. Figure 2-30 presents the results of an investigation into minimum allowable boom stiffnesses and boom buckling limits. It is apparent that the "V"-stiffened array does not have an optimum boom stiffness and blanket tension for a given frequency similar to the planar array. The following discussion further investigates these trends and has resulted in the selection of a minimum weight configuration.

2.5.3.3 V-Stiffened Array Computer Code

Tests on a planar array indicated that there were three regions of different stiffnesses for in-plane deflections as shown in Figure 2-31a and described

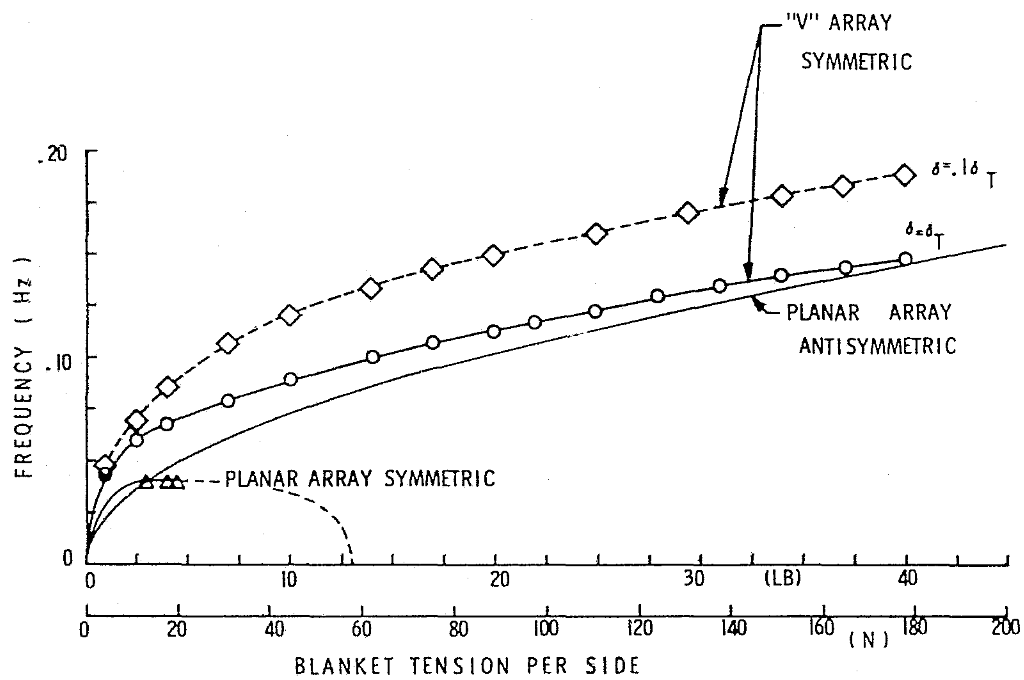


Figure 2-29 Effect of Blanket Tension on Solar Array Frequency

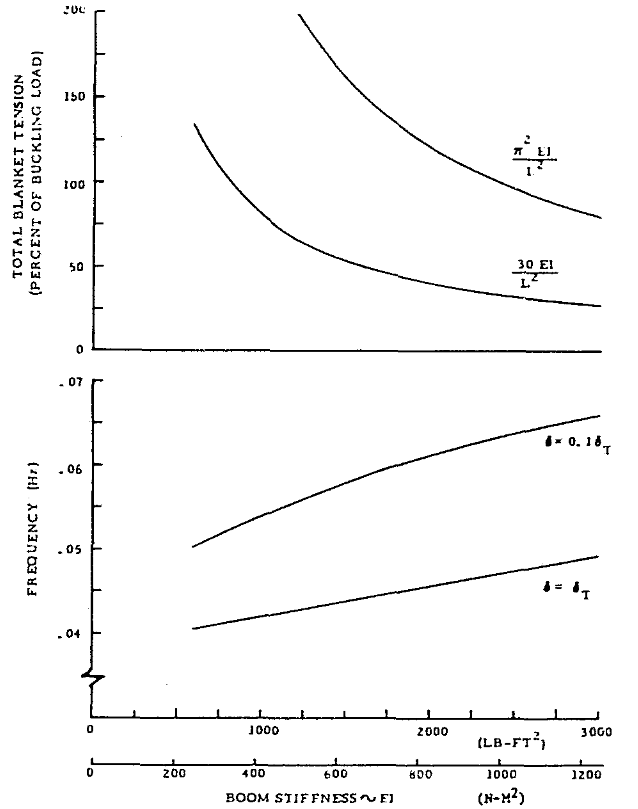
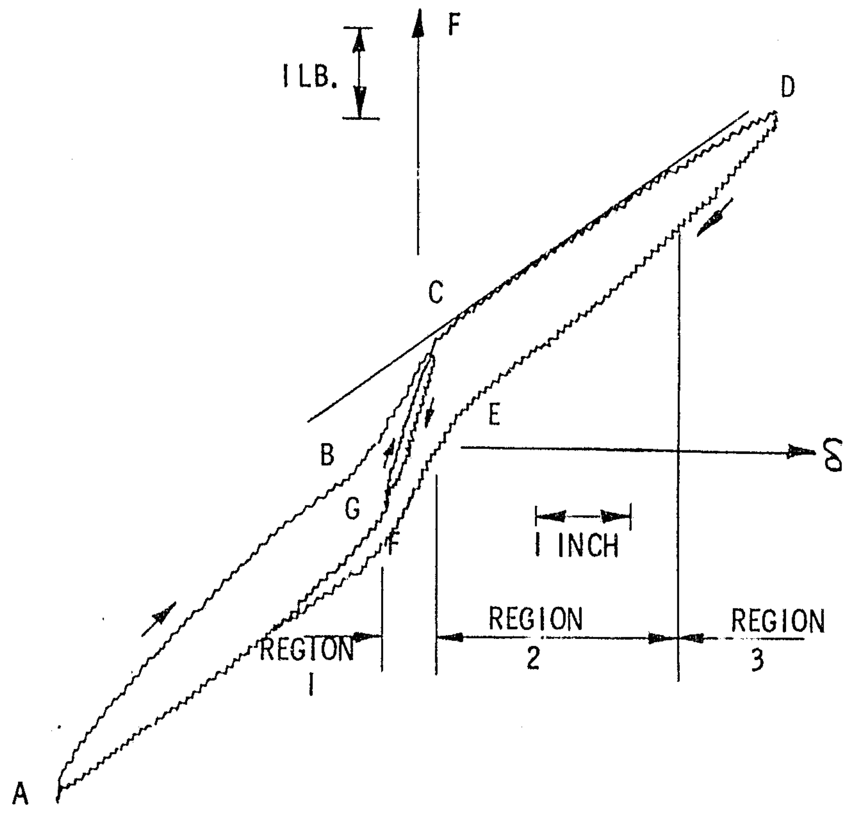
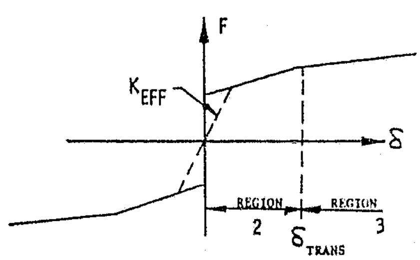


Figure 2-30 Effect of Boom Stiffness On "V" Configuration Solar Array Characteristics



(a) Measured on RA250



(b) Linearized

Figure 2-31 In-Plane Force-Deflection Characteristic

below:

Region 1: For small deflections, hysteretic behavior of the BI-STEM boom caused a relatively high stiffness. This is best predicted semi-empirically and is not predicted by simplified analytical modeling.

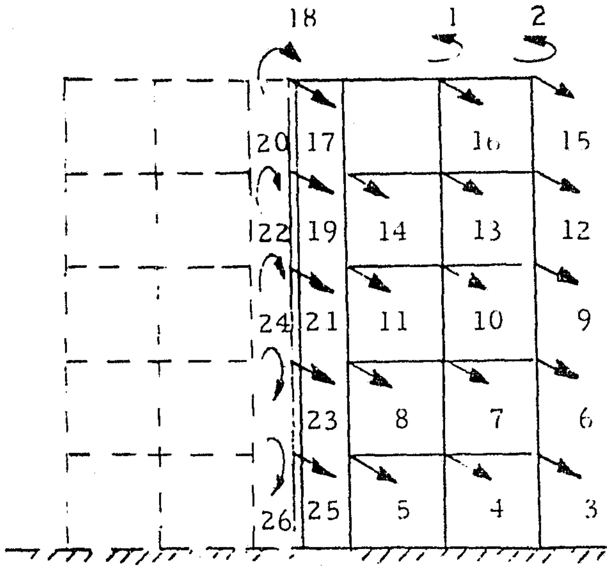
Region 2: For medium deflections, the tension distribution of the blankets changes such that the slope (θ) at the tip of the leading edge member is proportional to the tip deflection (δ) divided by the array length (L).

$$\theta = \delta / L$$

This results from a constraining moment at the tip of the boom due to the blanket tension and is valid until the tension shifts to the edges of the blanket.

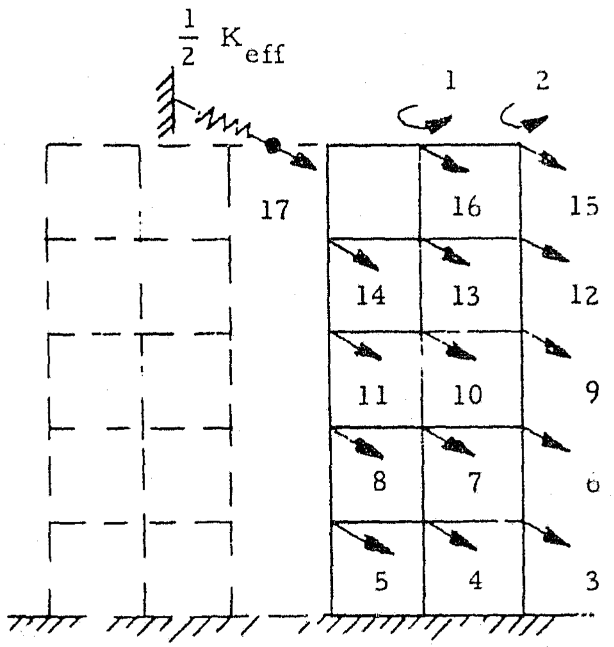
Region 3: For large deflections, the effect of blanket tension is no longer present and the boom behaves as a cantilever. This occurs after the transitional deflection (when tension shifts to the edges of the blankets at end of Region 2).

Using the idealized representation shown in Figure 2-31b, an effective linear stiffness (K_{eff}) can be defined for a selected amplitude of motion. Although other methods could be used to arrive at a linearized stiffness, this appears to be a reasonable estimate. It is conservative for large amplitudes in that the stiffness is higher than predicted, but may be unconservative for small amplitudes because the Region 1 stiffness is not included in the stiffness representation.



NOTE: SYMMETRY BOUNDRY CONDITIONS
IMPOSED ALONG BOOM ℄

(a) Planar Solar Array



NOTE: K_{eff} USED TO
REPRESENT BOOM
STIFFNESS WITH IN-
PLANE BLANKET
TENSION EFFECT.

(b) "V" - Configuration Solar Array

Figure 2-32 Symmetric Models

2.5.3.3.1 Symmetric Vibration Analysis

The approach used in performing this symmetric vibration analysis of the "V" stiffened array was to use the existing planar model of the blankets and revise the boom stiffness representation to reflect the effect of the blanket tension redistribution. The original analytical model of the array is shown in Figure 2-32a. The revised model of the array is shown in Figure 2-32b, where the major modification is to replace the boom finite element model by an effective linear spring (K_{eff}). This appears reasonable in that the cant angle of the array being considered is small (on the order 10^{o}) so that significant area is not added to the array due to the change in the projected area. The resulting change in the membrane stiffness due to the small angular rotation should not be significant, but should actually increase the blanket stiffness. Therefore, the main effect seems to be the revised boom stiffness.

Consider the out-of-plane deflections of the "V" stiffened array shown in Figure 2-33a and the free body diagram of the Leading Edge Member (LEM) shown in Figure 2-33b. These diagrams are identical to those of Reference 2 except for the modified width which now becomes the projected width; i.e., w is replaced by w $\sin\beta$. The force deflection relation (ref. 1) neglecting root flexibility ($\overline{\alpha}$) becomes:

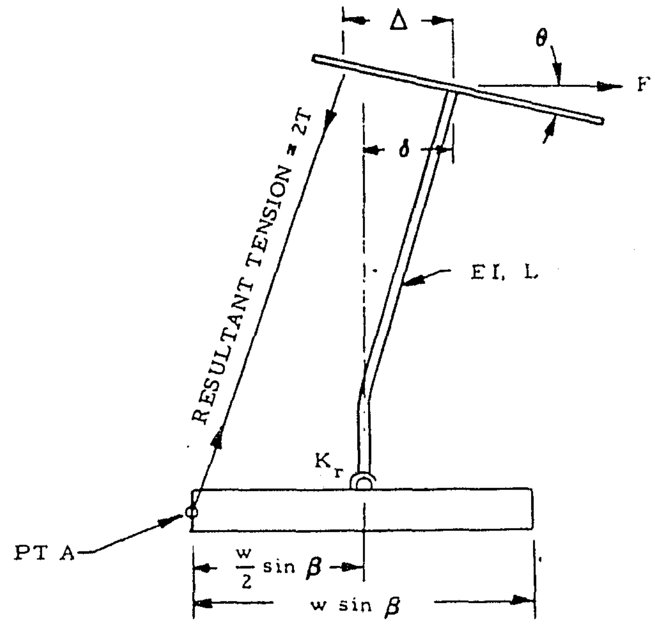
$$F = \left[\frac{4\,EI}{L^3} - \frac{4T}{15L}\right]\delta + \frac{Tw\,\sin\beta}{L} \qquad (1)$$

where:

T = tension per blanket

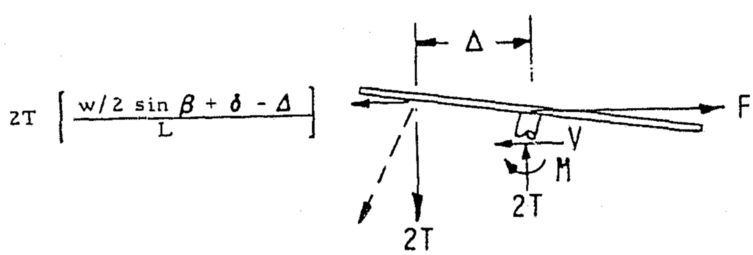
EI = Boom Stiffness

w = Half width of array



(a)

Edge View of Deflected Array



$$V = \frac{6\,E\,I\,\delta}{L^3\,(1+4\alpha)} - \left[\frac{11 + 76\alpha + 160\alpha^2}{5\,(1+4\alpha)^2}\right]\frac{T\delta}{L}$$

$$M = \frac{2\,E\,I\,\delta}{L^2\,(1+4\alpha)} + \frac{T\delta\;(1+12\alpha)}{15\,(1+4\alpha)^2}$$

(b)

Figure 2-33 Force Diagram at Outer End of Array

and the limiting deflection at which the transition from Region 2 to Region 3 occurs (Ref. 1 and 2) neglecting root flexibility, becomes:

$$\delta_{TRANS} = Tw \sin\beta \left[\frac{2\,EI}{L^2} + \frac{T}{15} \right]^{-1} \tag{2}$$

Using K_{eff} to linearize the boom stiffness over the range of applicable deflections and neglecting root flexibility:

$$K_{eff} = \frac{F}{\delta} = \frac{4\,EI}{L^3} - \frac{4T}{15L} + \frac{Tw \sin\beta}{L} \quad \frac{1}{\delta} \tag{3}$$

where 1/2 K_{eff} is added to the stiffness matrix of the analytical model at coordinate 17.

It will be noted that the tension effect on the boom stiffness is included in the linearized stiffness. As the tension is increased, the boom stiffness decreases as indicated by the first two terms of Equation 3. When $2T = \frac{30\,EI}{L^2}$, the boom stiffness becomes zero and the K_{eff} is due only to the initial offset value. From the buckling standpoint, the buckling load in Region 2 is increased from $\frac{\pi^2\,EI}{L^2}$ to $\frac{30\,EI}{L^2}$, an increase of approximately 3 to 1.

The mass of the boom is included in the boom tip coordinate using one-fourth of the boom mass.

2.5.3.4 Deployed Analysis

2.5.3.4.1 Assessment of Stiffening Effects

An evaluation of the effectiveness of the V-stiffened configuration was performed using the baseline configuration (Ref. 12) as a point of reference. The following values were assumed for the parameters specified:

Total Deployed Length	=	14 m (46 ft.)
Total Blanket Width	=	4.68 m (15.34 ft.)
Boom Stiffness (EI)	=	826 nt-m^2 (2000 lb.-$ft.^2$)
LEM Stiffness (EI)	=	2060 nt-m^2 (5000 lb.-$ft.^2$)
Blanket Mass	=	24.97 Kg (55.06/g_c slugs)
Tension per Blanket	=	5.78 nt (1.3 lbs.)

The effect of the cant angle on the array characteristics can be seen from the previous analytical expressions. The deflection at which transition occurs is directly proportional to the sine of the angle (Equation 2) so that the transitional deflection can be increased, if necessary, by increasing the angle (e.g., a 15 degree angle would result in approximately 50 percent increase in the transition deflection over that provided by a 10 degree angle). For a given deflection, the effective boom stiffness is increased significantly due to the increase in the offset force (3). On the other hand, the effective boom stiffness at the transitional deflection can be shown to be

$$K_{eff} = \frac{6\ EI}{L^3} - \frac{3T}{15L} \tag{4}$$

which is not affected by the cant angle.

The fundamental symmetric resonant frequency determined from the analytical model is shown in Figure 2-34 for the range of tension values investigated. For comparison, the symmetric and antisymmetric frequencies of the baseline planar array are also shown. The "V" array frequency is shown for oscillation amplitudes equal to the transitional deflection and one-tenth the transitional deflection. It should be noted that the small amplitude curve is questionable due to neglecting the Region 1 stiffness.

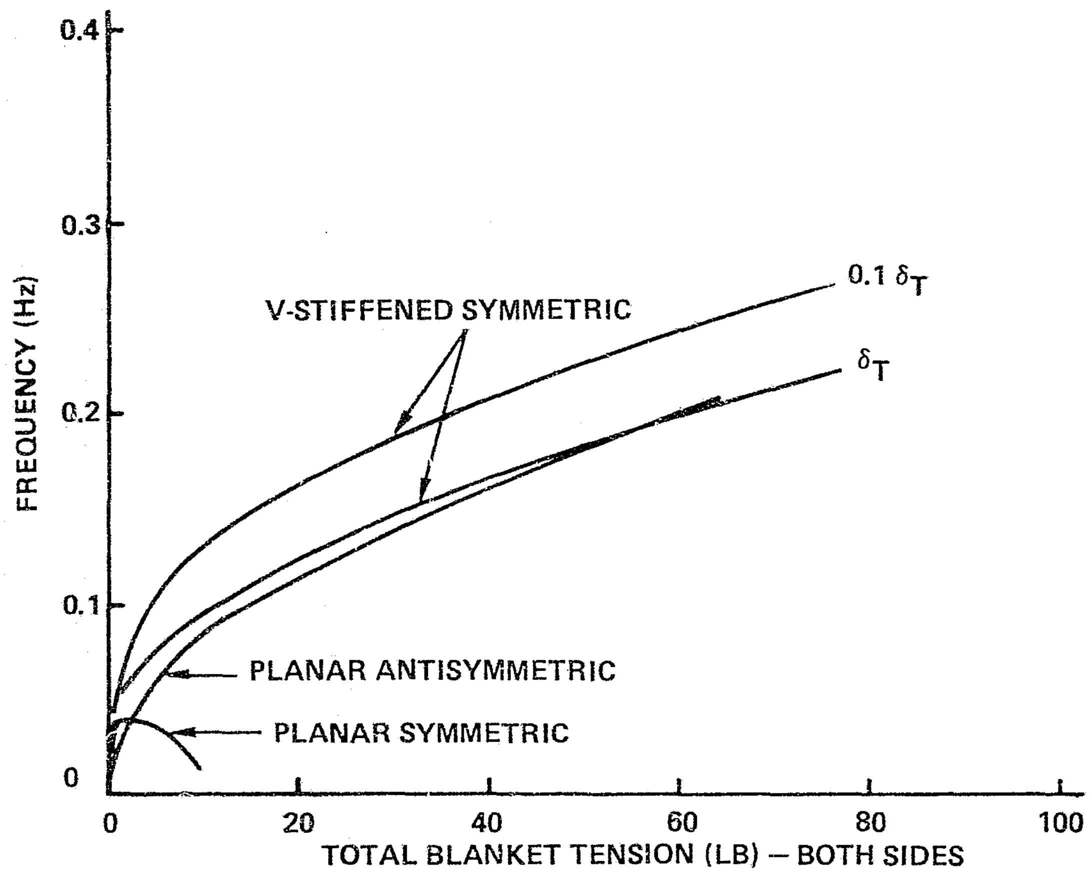


Figure 2-34 Effect of Blanket Tension on Solar Array Frequency

From Figure 2-34 it is apparent that the tension required to meet the .04 Hz torsional frequency and the boom stiffness required to meet the .04 Hz symmetric frequency criteria for the planar array, results in a greatly increased symmetric frequency for a small (10^{o}) amount of V-stiffening. Therefore, using this tension value (needed for .04 Hz torsion) allows the boom stiffness to be significantly decreased while still obtaining .04 Hz in out-of-plane bending.

For baseline design, the required blanket tension (per side) of a planar configuration was determined to be approximately 5.8 nt (1.2 lb). The tension was set at this value and the boom stiffness varied through the range of practical interest.

The range of boom stiffness that was considered practical was based on the buckling load of the array for the required tension value. If a conservative design approach is used, a criteria that the buckling load of the cantilever boom is not exceeded could be selected. Alternately, a less conservative criteria is that the buckling load of the boom with the blanket restoring moment acting would not be exceeded. Using the first criteria, the boom could not buckle for any range of deflections whereas the second criteria would result in boom buckling if the tip deflection was greater than the transitional deflection. From Figure 2-35 applying a safety factor of 1.25 to the buckling load an EI of 285 nt-m^2 (690 lb-ft^2) satisfies the first criteria and an EI of 91 nt-m^2 (220 lb-ft^2) satisfies the second criteria.

The calculated symmetric frequencies for this boom stiffness range satisfies the 0.04 Hz requirement for oscillations at the transitional deflection and a large margin is indicated for smaller oscillation amplitudes. Consequently, the controlling factor is boom buckling.

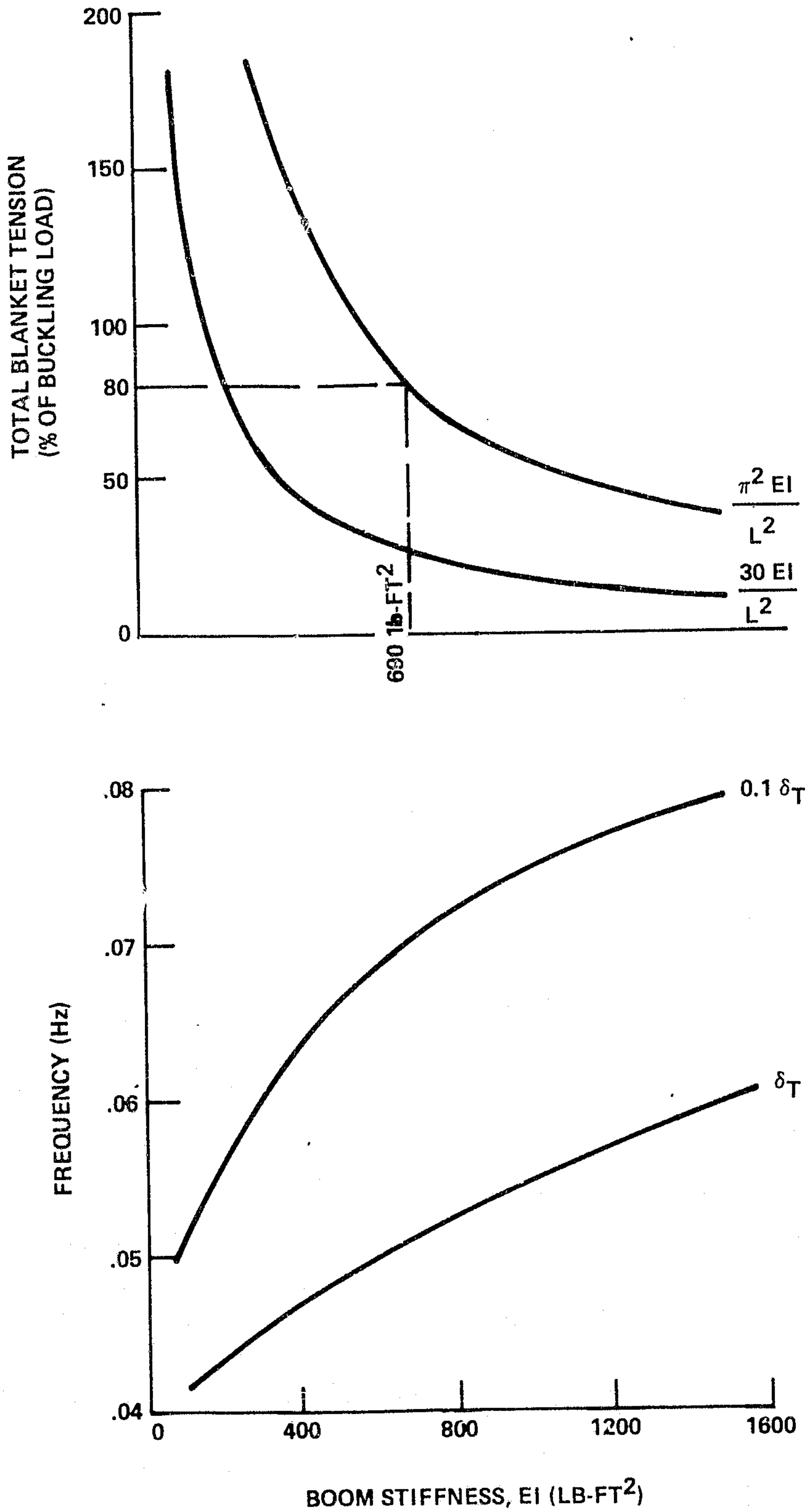


Figure 2-35 Effect of Boom Stiffness on "V" Configuration Solar Array Characteristics

2.5.3.4.2 Force Deflection Characteristics

To determine the optimum cant angle (β) of the blankets, computer runs were made to calculate the frequency in out-of-plane bending over a range of boom stiffnesses for cant angles of 6^o, 8^o, and 10^o. The transition forces (F_t) were calculated from equation (1) for each boom stiffness (EI) at each cant angle and are plotted vs. EI in Figure 2-36.

The loads produced by the quasi-static acceleration requirement (10^{-3} g's) set by JPL must not exceed the transition force or the effects of V-stiffening will be lost. A 1.25 safety factor applied to an approximate blanket plus boom weight of 57 lbs. multiplied by 10^{-3} results in a minimum transition force requirement of .071 lb.

The intersection of the horizontal line drawing at F_t=.07 on Figure 2-36 gives the minimum boom stiffness needed to meet this criteria for each of the cant angles. A blanket cant angle of approximately 8.25^o is seen to result in a design which meets exactly, both the transition force and boom buckling criteria. The use of any other cant angle would require a design which exceeds one or both of these requirements.

2.5.3.4.3 System Weight

In Figure 2-37 weights of the aritculated steel ASTROMAST and deployer, and the steel BI-STEM and deployer are plotted with blanket weight vs. blanket cant angle using the data shown in Figures 2-38, 2-39, 2-40 and 2-41. The boom stiffness used to determine these weights were the minimum required to meet both the transition force and boom buckling criteria as determined from Figure 2-36. The vertical line at 8.25^o indicates the point at which both the buckling and quasi-static loads criteria are met exactly. As can be seen from Figure 2-36

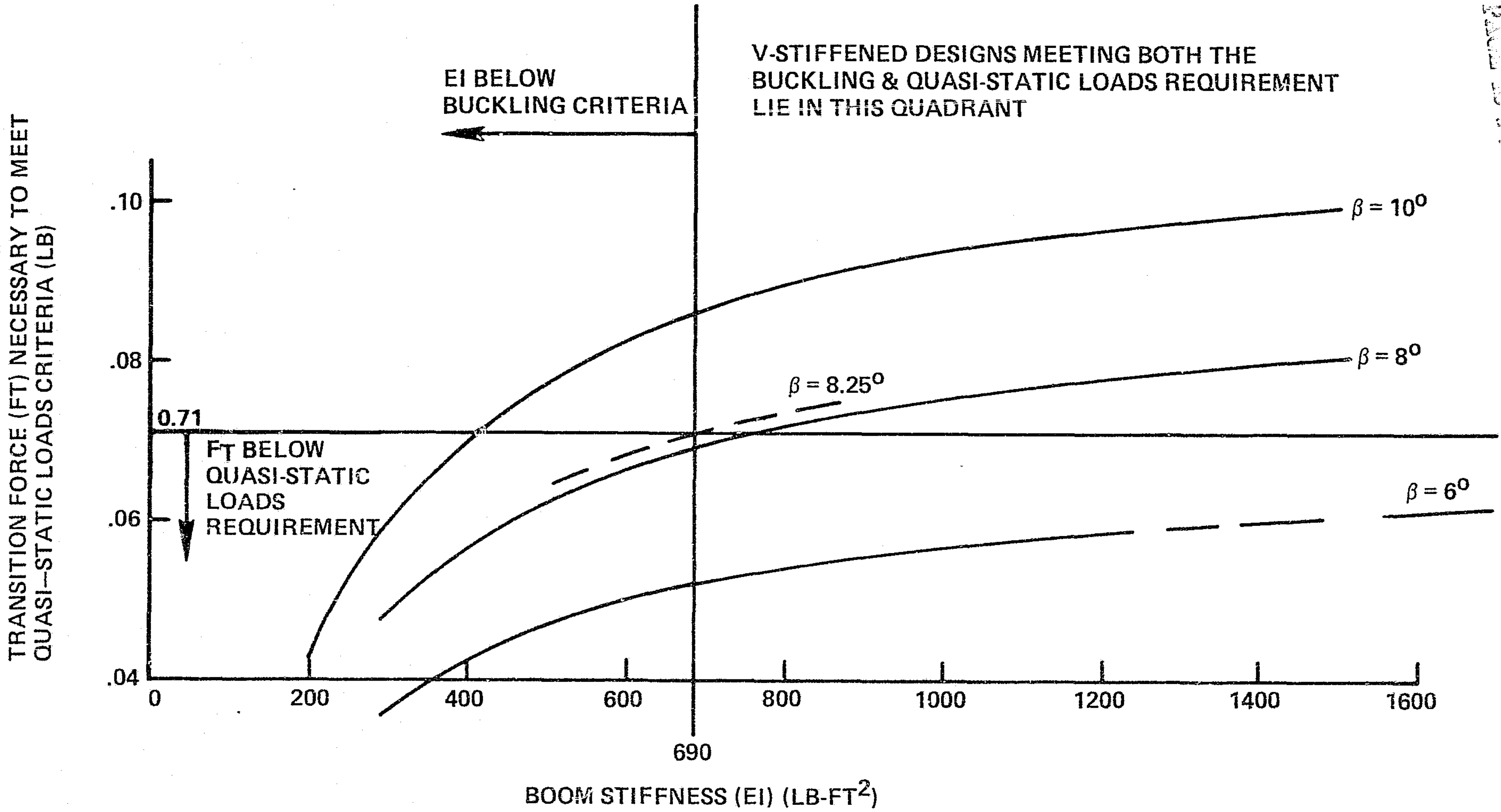


Figure 2-36 Effect of Blanket Cant Angle on Boom Stiffness

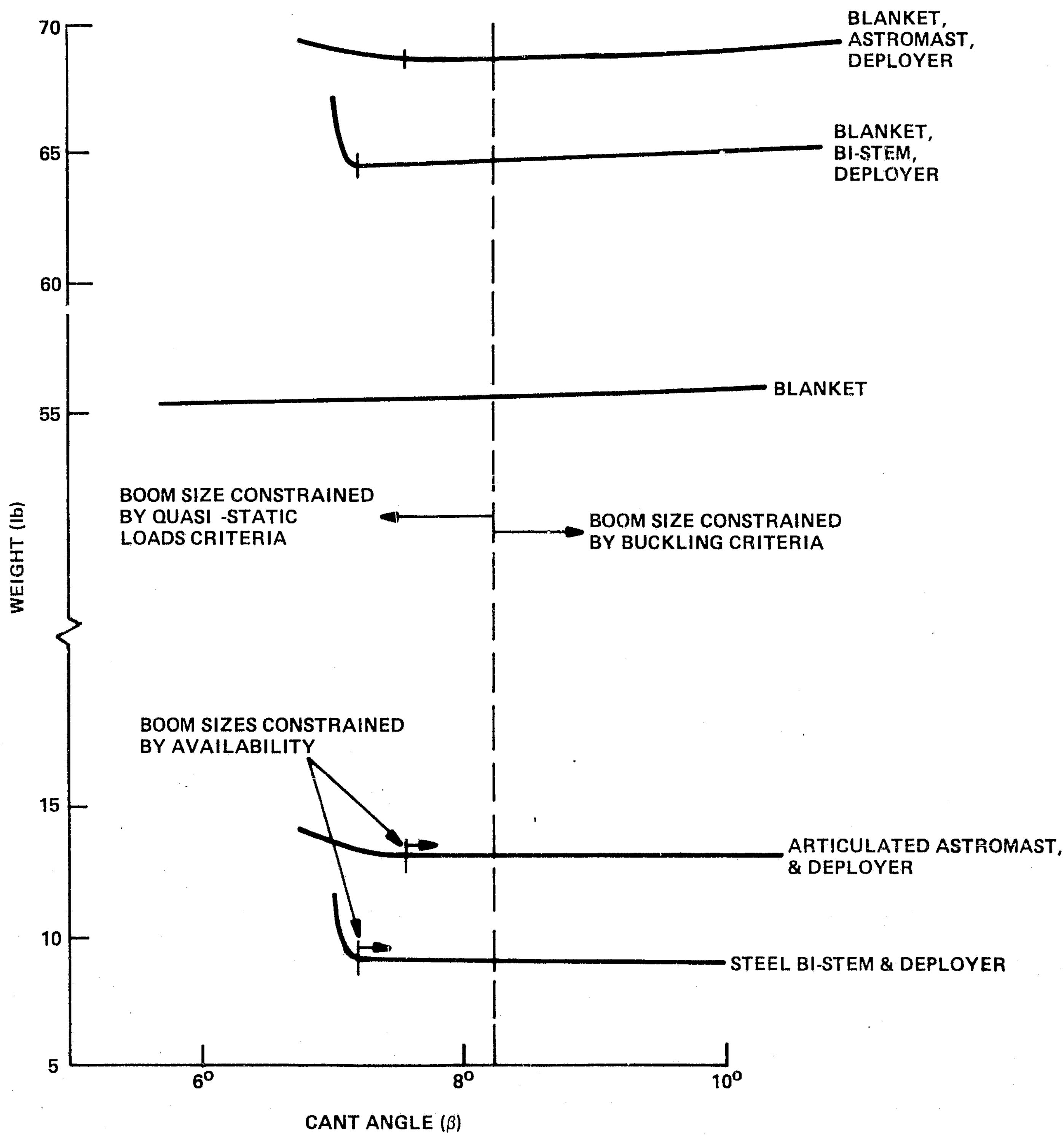


Figure 2-37 Effect of Blanket Cant Angle on Component Weights

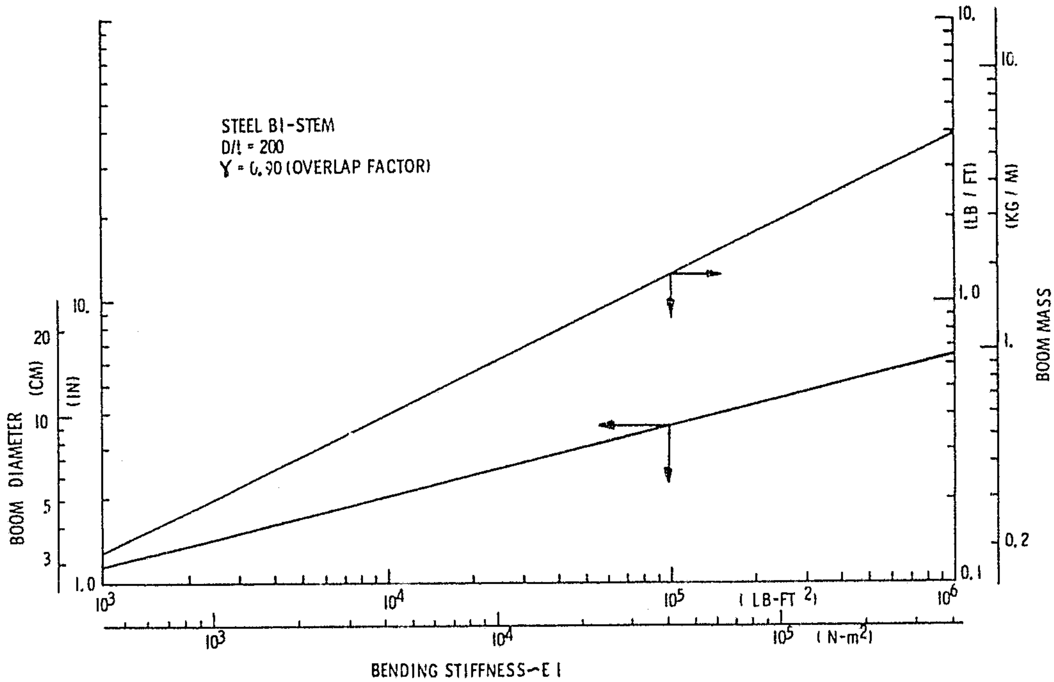


Figure 2-38 Steel BI-STEM Mass and Diameter as a Function of Bending Stiffness

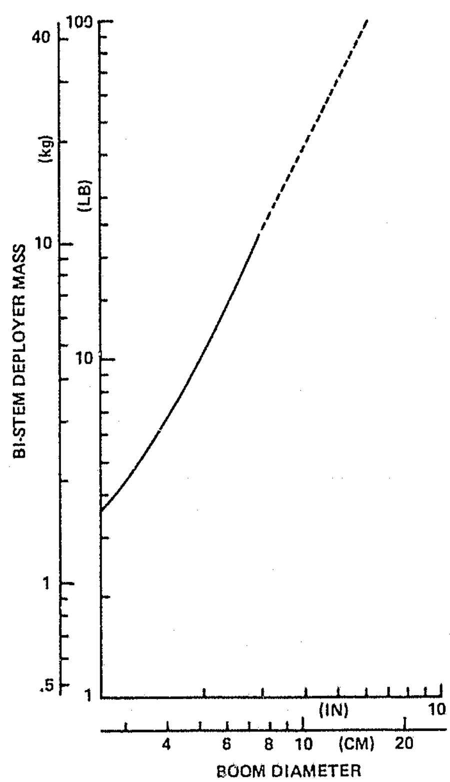


Figure 2-39 BI-STEM Deployer Mass Vs. Boom Diameter

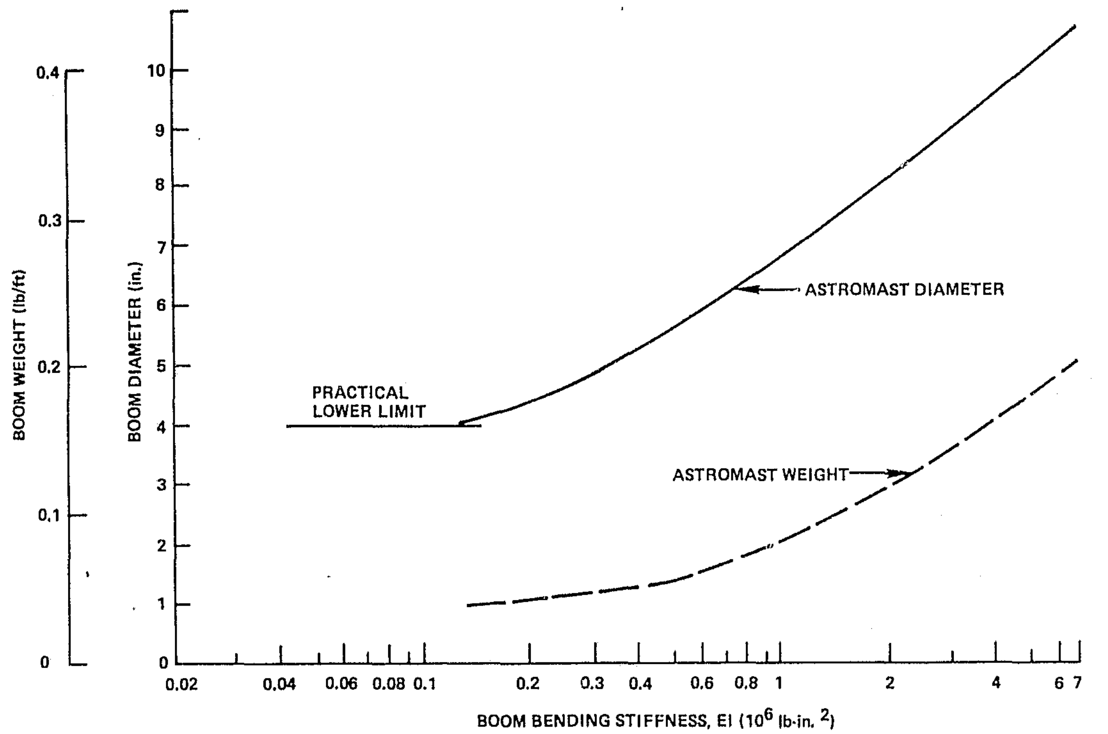


Figure 2-40 Dimensions and Weights of ASTROMAST Vs. Bending Stiffness (Astro Data)

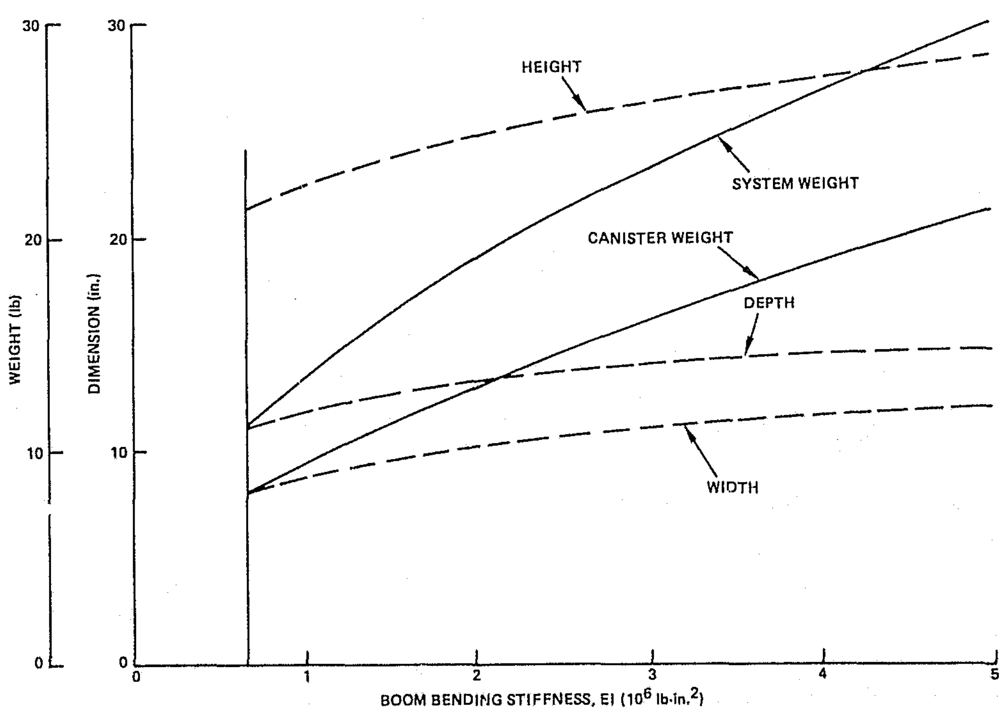


Figure 2-41 Weights and Dimensions of Canister Deployer for ASTROMAST

for angles greater than 8.25°, the boom size is constrained by the minimum EI of 690 $lb\text{-}ft^2$ needed to meet the buckling criteria, and at angles less than 8.25°, the boom size is constrained by the EI needed to meet the quasi-static loads requirement.

In determining these weights it has been assumed that the smallest boom represented on the plots (Figures 2-38, 2-39, and 2-40) is the smallest which can be made. This results in the boom and deployer weights being constant beyond 7.2° of cant angle for the BI-STEM and beyond 7.5° for the ASTROMAST. ASTROMAST has indicated that it is unlikely that their boom can be obtained in sizes smaller than that given in Figures 2-39 and 2-40. It is not known at this time whether the steel BI-STEM can be obtained in smaller sizes. Should they become available, it could result in a significant weight savings over that shown for the BI-STEM in Figure 2-37.

2.5.4 COMPARISON SUMMARY

Table 2-15 presents a summary comparison of the 110 W/kg and 200 W/kg baseline arrays established by the previously described analysis techniques. The effect of using lighter, more efficient solar cells is immediately obvious in the sizeable reduction of blanket weight and this is further reflected in the lower boom bending stiffness and deployment mechanism weights. However, the significant reduction in required boom bending stiffness for the "V" stiffened array is mostly dissipated by the practical size limitations of both the ASTROMAST AND BI-STEM hardware. The values given for the 200 W/kg array are for the smallest units listed in the vendor literature, namely a 4 inch ASTROMAST and 1.3 inch BI-STEM boom. Unfortunately, a point of diminishing return has been reached with the deployer where practical minimum size limitations negates the advantages of smaller and lighter weight booms.

In view of the above noted limitations, several alternatives are suggested in order to achieve further optimization in the design. The simplest approach is to consider larger power arrays than the present 10 kilowatts. By going to a larger array size the practical limitations of the hardware are avoided while the same optimized configuration parameters are maintained. The optimization of the stowage and support structure has yet to be investigated but larger size hardware can provide for more efficient utilization of structure. And finally, the validity of the blanket weight and 3 mil solar cells has to be confirmed. If this is found to be unattainable, then the deployment hardware would have to be proportionately increased in size to meet the increased blanket weight requirements.

2.5.5 DISCUSSION OF RESULTS

The results of the above analyses indicate a possible trend in the 200 W/kg proposed concepts. For instance the analysis shows that a bending stiffness of 1×10^5 lb.-in.^{2} is adequate for the array. The ASTROMAST and BI-STEM stiffness versus diameter characteristics are shown in Figures 2-40 and 2-42 respectively. From this data it can be seen that the minimum practical ASTROMAST size (4-inch diameter) more than meets the stiffness requirement. The curves show that a 0.9-inch diameter BI-STEM would just meet the 1×10^5 lb-in^2 requirement. However, since 0.9-inch is a non-standard size, the next size larger, 1.34-inch diameter, would be a more practical choice. It is also of interest to note that the 1.34-inch diameter BI-STEM would also meet the planar EI requirement of 3×10^5 lb-in^2, thus creating more flexibility in the choice between planar and V-stiffened configurations.

Using this information on appropriate mast size, two preliminary mechanical concepts have been generated to stimulated thinking on the design problems

Table 2-15
Comparison of 110 W/Kg and 200 W/Kg Baseline Arrays

Array		Cant	Boom	EI (nt-m^2/lb-in^2)	Wt. (Kg/lb)		
					Blanket	Boom	Deployer
110 W/kg	Planar	0°	Articulated Steel Astromast	4170/14 x 10^5	48.5/107	3.1/6.8	5.3/11.7
	V-Stiffened	10°	Steel BI-STEM	1230/4.3 x 10^5	49.2/108.5	4.4/9.7	2.5/5.5
200 W/kg	Planar	0°	Articulated Steel Astromast	826/3 x 10^5	256/56.3	.82/1.8	5.1/11.3
	V-Stiffened	8°	Steel BI-STEM	290/1 x 10^5	25.8/56.9	2.1/4.6	2.0/4.4

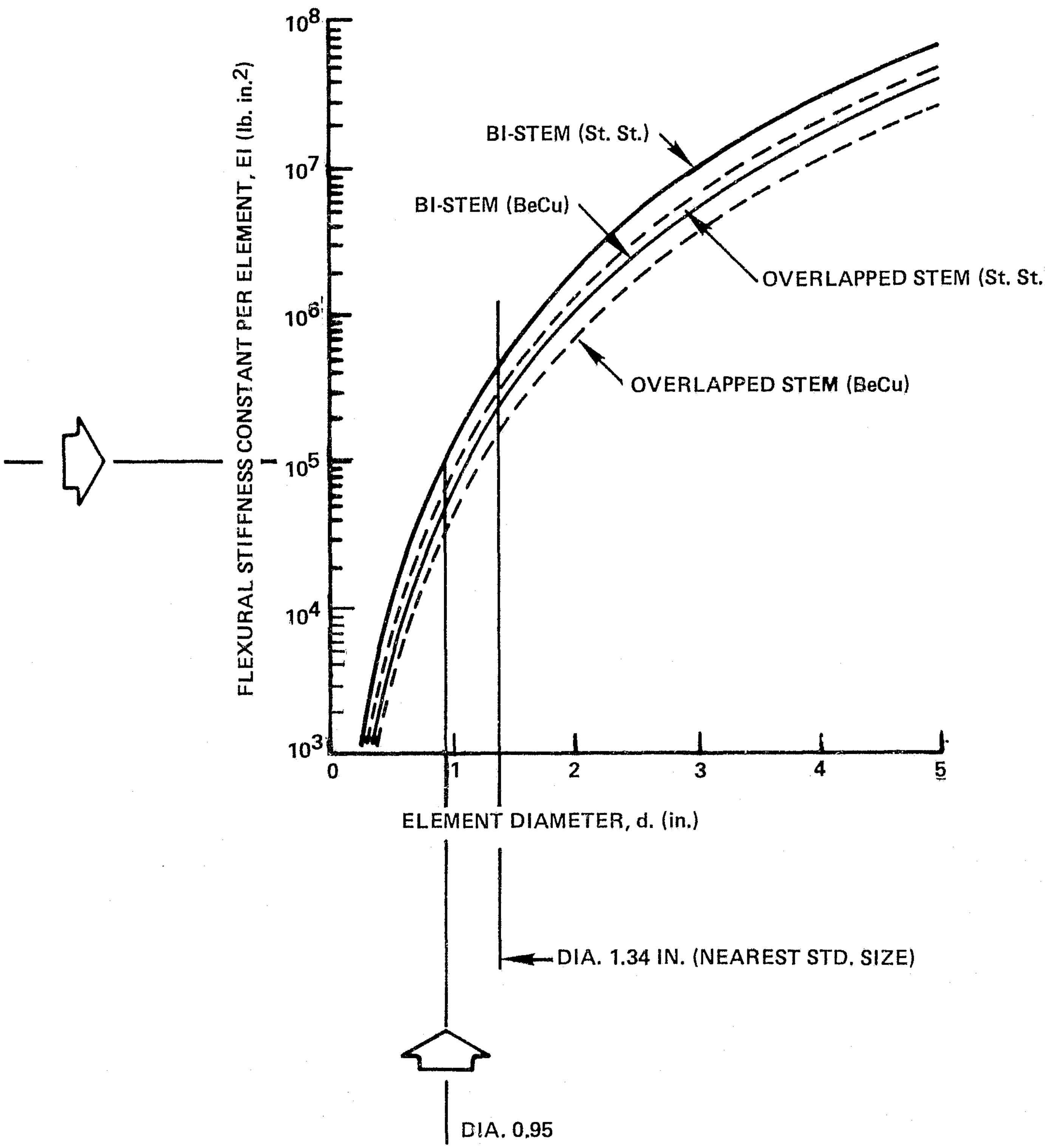


Figure 2-42 BI-STEM Diameter Requirements

posed by the 200 W/kg array. A rollup planar array is illustrated in Figure 2-43. This concept shows the general appearance of a 4 sided drum of 12-inch radius at the corners, and the use of a 4-inch diameter Astromast. The concept of V-stiffening is shown in Figure 2-44. A 16-inch diameter stowage drum is tentatively sketched in position for purposes of illustration.

Possible weight benefits to be derived from present concepts are shown the "Mass Summary Chart", Table 2-16. The weight values shown on this chart for new concepts are based on preliminary estimates and extrapolations. They are, of course, subject to change up or down as the study becomes more detailed. However, they do serve to show an estimate of the degree of progress in weight optimization. The chart will be verified and extended as the study progresses.

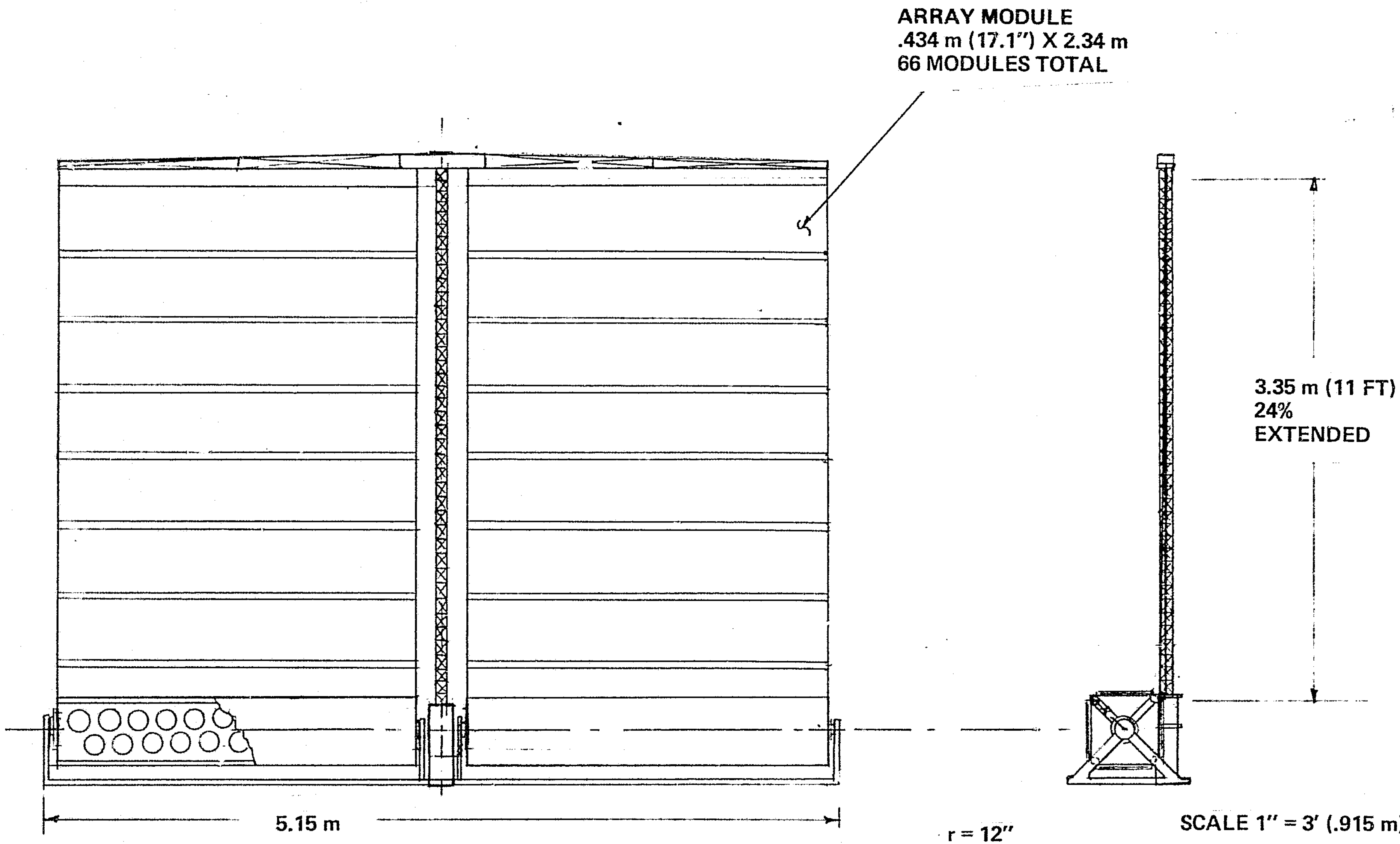


Figure 2-43 200 W/kg Solar Array Flat Sided Roll Up Concept

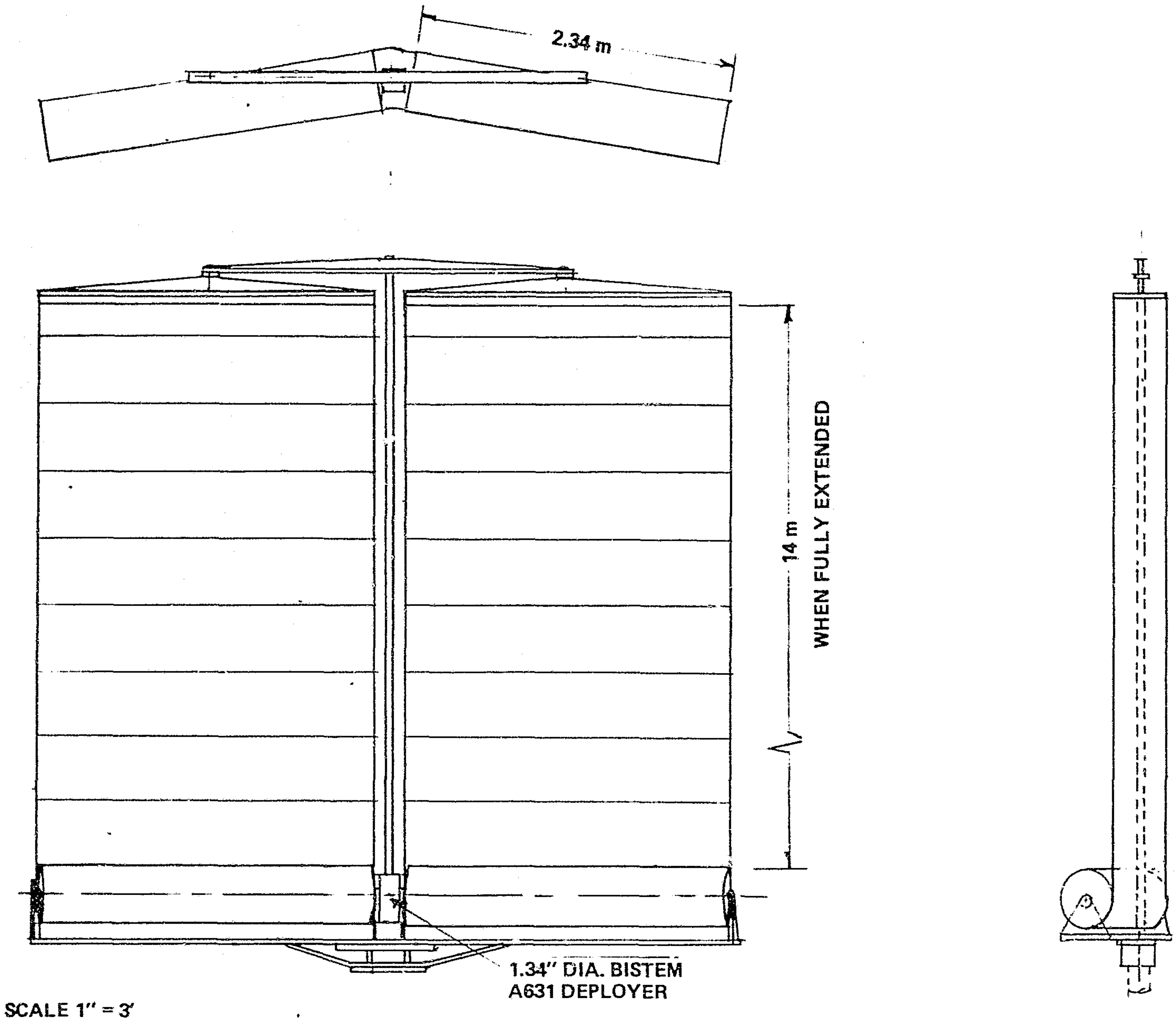


Figure 2-44 V-Stiffened Concept 200 Watt/Kg Array

C-2

Table 2-16 Mass Summary Chart

200 WATT/KG SOLAR ARRAY STUDY

ITEM			ESTIMATED WEIGHT					
	STATE OF ART (SEPS ARRAY)		110 W/KG ARRAY GE STUDY		200 WATT/KG CONFIGURATIONS			
					PLANAR/ASTROMAST		V-STIFFENED/BISTEM	
	WT. IN KG	%	WT. IN KG	%	WT. IN KG	%	WT. IN KG	%
	②				①		③	
Electrical								
Solar Cells			29.26		11.91		11.91	
Substrate			4.04		2.51		2.51	
Adhesive			3.50		2.18		2.18	
Cover Material			7.26		4.51		4.51	
Bus Strips & Insulators			1.50		1.50		1.50	
Inboard & Outboard Leaders			.27		.27		.27	
Interconnectors, Terminals			1.96		1.96		1.96	
Strip Hinge Joints & Jumpers			.71		.71		.71	
Slip Ring/Flex Lead			-		-		-	
Sub Total	122.45	64	48.5	55	25.55	41	25.55	48
Mechanical								
Stowage & Support								
Frame								
Container								
Cover								
Drum & End Plates								
Cushion								
Sub Total	25.88	13	30.6	35	24.21	39	18.25	35
Retention & Release Mech								
Brackets								
Cables								
Pyro Devices								
Sub Total					2.4	4	2.4	5
Deploy & Retract Mechanis,								
Mast	35.05		3.1		.82		2.1	
Canister/Deployer			5.3		5.10		2.49	
Drum Actuator (Spring					.59		.59	
Springs (Retention)	8.91							
Sub Total	43.96	23	8.4	10	6.51	16	5.18	12
Total	192.29	100	87.5	100	58.67	100	51.38	100
Efficiency (Watts/KG)	65		114.3		170.4		194.6	

NOTES

① Extrapolation of 100 W/Kg Baseline design. (4" Dia. Astromast)
② SEPS basline design by LMSC.
③ Low wt drum & 1.34" Dia Bistem A631 deployer.

SECTION 3

CONCLUSIONS

No definite conclusions can be made at this early stage of the program. However, some preliminary observations seem noteworthy.

The use of 3 mil solar cells has a weight advantage that overshadows the reduction in power output as described in paragraph 2.4.2 of this report. Additional evaluations will be performed to determine the practicality of manufacturing, handling, and assembling such thin silicon cells. It may be necessary to consider thicker cells (and their associated increased weight) in order to establish confidence in a practical approach.

The use of the V-stiffened array reduces the stiffness required for the boom over that required for a planar array. The reduced stiffness allows a smaller and lighter deployment mechanism and boom to be used to meet the 0.04 Hertz minimum natural frequency.

SECTION 4

RECOMMENDATIONS

No specific recommendations can be made at this early stage of the program.

SECTION 5

NEW TECHNOLOGY

No items of new technology have been reported during this period.

SECTION 6

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Document #200 W/Kg-2.76-004 Rev. 1
15 April 1976

Baseline Requirements

for a

200 Watt/Kilogram Lightweight

Solar Panel Subsystem

Contract No. 954393

Prepared by Kevin M. Speight
Kevin M. Speight, Proj. Mgr.
200 W/Kg Solar Array Study

1.0 SCOPE

1.1 This specification covers the requirements for a conceptual approach for a 10 kilowatt solar panel design having a power-to-weight ratio of 200 watts per kilogram or greater. This conceptual approach requires a background of information on the influencing parameters, their margins, the trade-offs considered, and the rationale developed for a light-weight array design as defined by the requirements in paragraph 3.0.

2.0 APPLICABLE DOCUMENT

2.1 The following document forms a part of this specification to the extent specified herein:

MIL-HDBK-5	Metallic Materials and Elements for Flight Vehicle Structures

3.0 REQUIREMENTS

3.1 <u>Conflicting requirements.</u> In case of conflict between the requirements of this specification and the documents referenced herein, the requirements of this specification shall govern.

3.1.1 <u>Deviations from standard practices.</u> Any deviations from generally accepted standard practices will be approved by the Jet Propulsion Laboratory (JPL), after it has been demonstrated by analysis that the deviations will not degrade the overall probability of attaining the objectives of this effort. The burden of proof in such circumstances shall rest

upon the contractor and not upon JPL.

3.2 Performance requirements. The solar panel shall be designed so that the following performance requirements can be met.

3.2.1 General. In the stowed configuration, the solar panel shall be supported in a manner that will prevent damage to the solar panel under shock and vibration loads. Upon command and in proper sequence, the release and deployment mechanism shall extend and lock the solar panel into the deployed position at a rate to be defined by the contractor. Upon command and in proper sequence, the retraction mechanism shall retract up to 90% of the solar panel, exposing sufficient area to provide up to 10% of the total power, and lock to this partially stowed position at a rate defined by the contractor. This retraction mechanism will be considered as an option.

3.2.2 Power requirement. Following launch, the deployed solar panel shall be capable of supplying 10 kilowatts of electrical power at the spacecraft interface at a solar intensity normally incident at 1 AU* and at the predicted solar array temperature at this intensity.

3.2.3 Lifetime. The solar panel shall be designed to perform over a period of 3 years with no greater than a 20 percent loss of power, disregarding solar flare proton reduction, and with no failures which sould prevent the panel from performing

*1 AU is defined in ASTM Spec E490-73A

successfully in both electrical and mechanical modes. Sound engineering judgement shall be exercised in regard to the depth to which the design is driven by the exclusion of single or multiple failure modes.

3.2.4 Solar panel operating temperature. The thermal characteristics of the deployed panel shall be adjusted so that the celled area maintains a maximum operating temperature of 85°C at a solar intensity normally incident at 1 AU*. The electrical characteristics of the array shall be determined over the temperature range of -100°C to +100°C.

3.2.5 Solar panel weight. The weight of the solar panel in flight configurations, including the release, deployment, and retraction mechanisms, but not including solar panel gimballing mechanisms, shall be such that the solar panel specific power equals or exceeds 200 watts per kilogram at a solar intensity normally incident at 1 AU*.

3.2.6 Packaging volume envelope. The volume and shape of the stowed solar panel, including the release, deployment, retraction and lock (an option) mechanisms, shall be determined by the contractor in order to maximize the solar panel adaptability to various spacecraft configurations. In these design considerations, a 2000-pound spacecraft (which includes two 10-kilowatt solar panels and a Shuttle launch vehicle) shall be assumed.

*1 AU is defined in ASTM Spec E490-73A

The following requirements shall also be included:

a) Launch vehicle shroud volume restrictions

b) Spacecraft structural interface requirements

c) Solar panel deployment complexity (reliability)

d) Solar panel gimballing (Sun tracking) requirements

e) Solar panel retraction complexity (reliability)

f) Solar panel attachment configuration requirements.

3.2.7 Structural interfaces. The solar panel to spacecraft attachment points shall be considered to provide the most efficient interface capable of performing the mission. Consideration shall be given to the ease with which the deployed solar panel can be gimballed (tilted or rotated) with respect to the spacecraft as required by the Sun tracking requirements. Consideration shall also be given to the requirements imposed on the spacecraft structure by the solar panel. A solar panel requiring an extremely rigid support or negligible relative motion between widely spaced support points is undesirable because meeting these requirements might result in increased spacecraft weight.

3.2.8 Structural rigidity. In the deployed configuration, the solar panel shall have sufficient rigidity so that its lowest cantilevered natural frequency of vibration is equal to or greater than 0.04 Hz. In the event this criteria cannot be met, i.e., the cantilevered natural frequency is less than 0.04 Hz, the

interaction of the flexibility of the solar array to the JPL attitude control system shall be analyzed to assess the impact of the flexibility of the solar panels on the attitude control systems.

3.2.9 Mass center location. The solar panel shall be designed to minimize displacement of the vehicle mass center and center of solar pressure caused by thermal gradients and solar panel temperatures.

3.2.10 Flatness. In the deployed configuration the solar panel blanket celled area shall lie in a predetermined plane with a maximum angular deviation from this plane of ten (10) degrees. This deviation shall include deflections caused by thermal gradients but shall not include deflections caused by dynamic mechanical load inputs.

3.2.11 Inspection. Release, deployment, retraction, and locking mechanisms, shall be designed so that, with suitable ground support equipment, their operating functions can be inspected in a one-g Earth field environment prior to installation on the spacecraft.

3.2.12 Reliability. The solar panel design shall incorporate design practices that enhance the probability that the solar panel will operate successfully in both mechanical and electrical modes.

3.3 Environmental requirements. The following environmental requirements shall be considered in the design of the solar panel.

3.3.1 Ground handling. The solar panel's structural, mechanical, and electrical performance shall not be degraded because of ground handling during manufacturing, testing, and transportation operations.

3.3.2 Launch environment. The following environmental constraints which represent the launch environment of the solar panel in the stowed configuration, shall be considered in the solar panel design.

3.3.2.1 Sinusoidal vibration. Sinusoidal vibration input levels as shown below will be applied at spacecraft solar array interface in three orthogonal directions, at a sweep rate of one octave per minute.

Frequency Range (Hz)	Amplitude
2-5	1.0-inch double amplitude
5-26	1.3 g (0-pk)
26-50	0.036 inch double amplitude
50-1000	5g (0-pk)

3.3.2.2 Acoustic. The launch acoustics environment shall be 60 seconds of a random incidence, reverberant sound field, having the third-octove band sound pressure levels defined in Fig. 1.

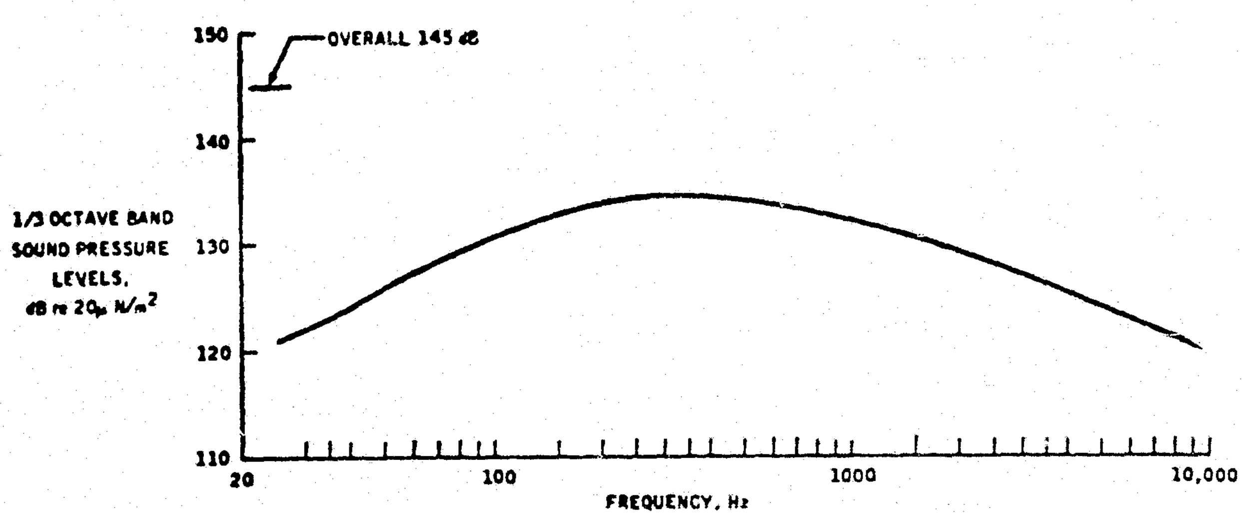


Figure 1. Acoustic Test Levels

3.3.2.3 <u>Shock.</u> The worst shock environment will be a 20 g terminal sawtooth shock pulse of 10 milliseconds duration at the spacecraft interface in each of three orthogonal directions.

3.3.2.4 <u>Static acceleration.</u> The static acceleration environments shall be 9 g's at the approximate center of mass of the solar panel in the stowed configuration. This environment shall be considered for the axial axes; 2 g's shall be considered for the lateral axis.

3.3.2.5 <u>Launch pressure profile.</u> The solar panel temperature shall be initially at 27 $\pm 6^{o}$C and at atmospheric pressure. Figure 2 shows the pressure-time history during launch and ascent.

3.3.3 <u>Space flight environment.</u> The following space flight environmental constraints shall be considered in the solar panel design.

3.3.3.1 <u>Steady state thermal/vacuum environment.</u> The steady state thermal/vacuum environment shall cover the range from -130 to +140^{o}C and a pressure of 10^{-5} torr or less.

3.3.3.2 <u>Thermal shock environment.</u> The thermal shock temperature extremes shall be -1[illegible]^{o}C and +140^{o}C at a pressure of 10^{-5} torr or less. The temperature time rate of change during thermal shock shall be the natural cooling rate of the solar panel in a simulated passage into a planetary shadow with an assumed planetary albedo of zero, and the natural heating rate of the solar panel in a simulated passage from a planetary shadow into a normal solar

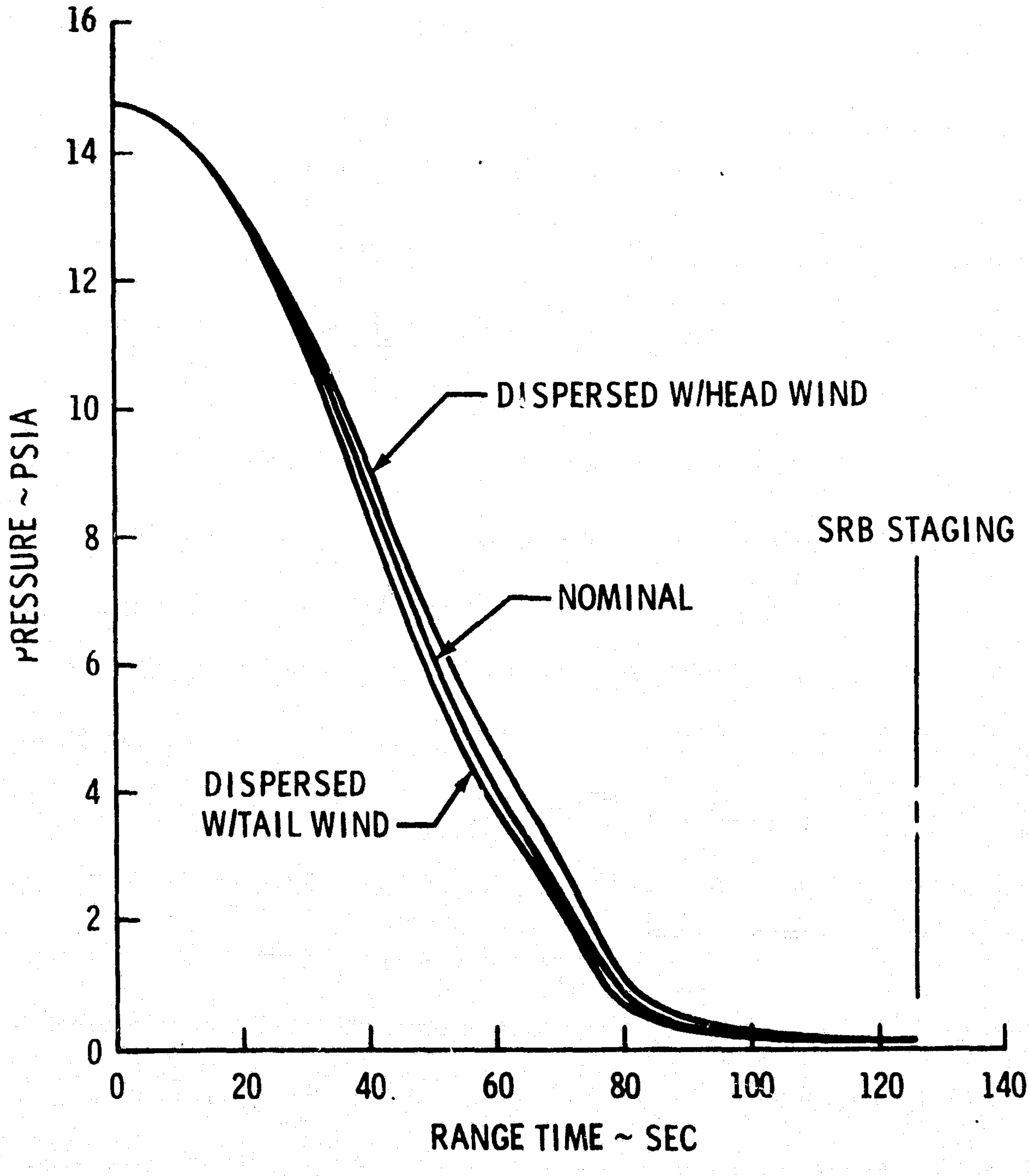


FIGURE 2. ORBITER PAYLOAD BAY INTERNAL PRESSURE HISTORIES DURING ASCENT

flux of intensity corresponding to a steady state temperature of 140°C on the solar cells. The total thermal shock environment shall consist of 1000 complete cooling and heating cycles.

3.3.3.3 Solar flare proton radiation environment. The proton fluency for the 3-year mission shall be as defined in Table 1.

Table 1. Mission Proton Fluency

Proton Energy (Mev)	Total Fluency (Particles/cm^2)
1	2.0×10^{12}

3.3.3.4 Pyrotechnic shock environment. The solar panel assembly shall be capable of withstanding shock environments induced by the firing of any pyrotechnics that may be required for the operation of the assembly.

3.4 Materials, parts, and processes. Materials, parts, and processes used in the design of the solar panel shall conform to the requirements specified herein. Any materials, parts, and processes that are not so covered shall be subject to the approval of the JPL cognizant engineer. In every case, the contractor's selection shall assure the highest uniform quality of the solar panel.

3.4.1 Material selection criteria. The influence of the following environments and those specified in 3.3 on the design properties

of the structural, electrical, thermal control, and lubricant materials in the solar panel shall be considered:

(a) Storage at 95 percent relative humidity at 55°C for 50 hours. The solar array may, however, be protected during delivery to the launch facility by use of appropriate site or ground operations facilites or equipment. If such protection is deemed appropriate, cost weight, and other impacts on the array design shall be evaluated.

(b) 10,000 thermal cycles between -190°C and +140°C at 10^{-7} torr with a 90-minute cycle, and a temperature stabilization ($< 2°C/hr$) dwell at the extreme temperatures.

(c) 1000 thermal shocks as defined in para. 3.3.3.2 "Thermal Shock Environment".

3.4.1.1 Flight environment materials. The materials shall be capable of enduring the space environment without releasing any significant condensing gases which would decrease the solar cell efficiency, or could potentially lead to electrical shorts or degradation to the spacecraft system operation.

3.4.2 Radiation resistance. The dosage and energy levels of the particulate radiation encountered during a mission shall not produce a significant effect on the metallic structural elements. Polymeric materials shall be either shielded or selected to resist a radiation surface dosage of 10^7 rads without decreasing the critical design properties below the design allowables.

3.4.3 Exposed structural adhesives. When adhesives are considered for bonding transparent or partially transparent structural components, the influence of particulate surface dosage radiation of 10^7 rads, and ultraviolet radiation equal to 1095 days of solar radiation at the rate of 2.002 calories/cm^2/minute, on the adhesive shall be considered.

3.4.4 Diodes. Diode isolation will be provided in the Power Conditioning circuitry. Therefore, diodes are not required on the solar array.

3.4.5 Solar Cells. The candidate solar cells to be used have the following characteristics:

(a) Current-voltage temperature coefficients between $-100^{\circ}C$ and $+100^{\circ}C$ at 1 AU:

Current: 0.03 ma/$^{\circ}$C-Cm^2
Voltage: -2.0 mv/$^{\circ}$C

These values apply to all cells of any thickness between 0.003 and 0.010 inch.

(b) Physical properties:

Length: 2 to 4 Cm
Width: 2 Cm
Thickness: 0.025 Cm (.010 inch) to 0.0075 Cm (0.003 inch)

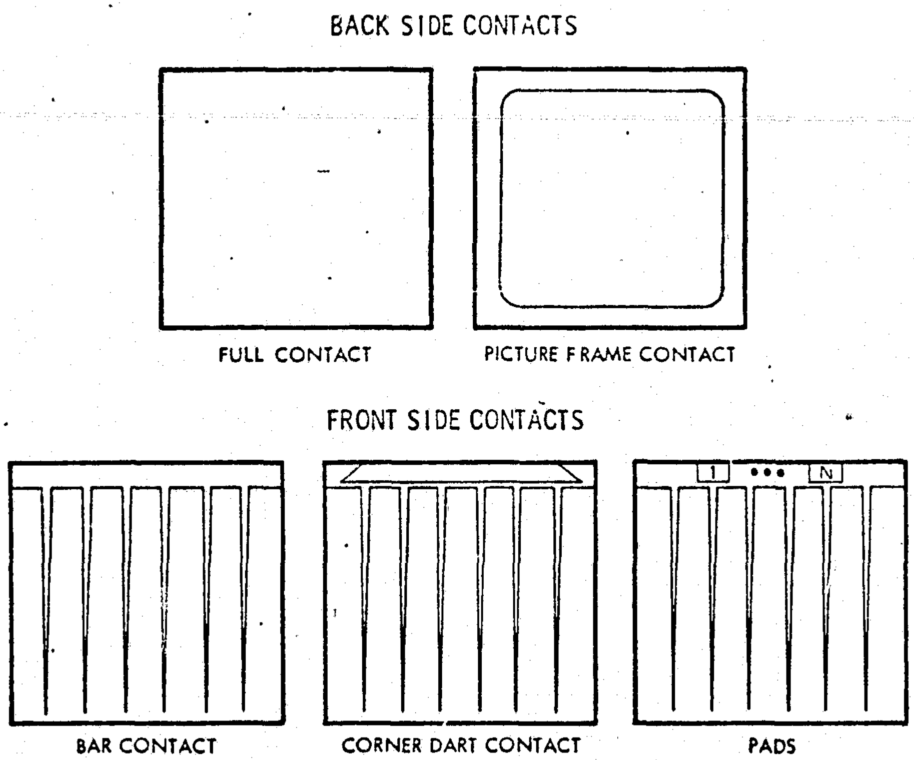
(c) Practical contact configuration. See Figure 3.

(d) Interconnecting methods:

Both weldable and solderable solar cells will be considered as available for cell thicknesses between 0.003 and 0.010 inch. The cell contacts will be silver-palladium-titanium

FIGURE 3

SOLAR CELL CONTACT CONFIGURATION



BAR CONTACT:	FULL WIDTH EITHER EDGE, 20 MILS TO ANYTHING DESIRED
WRAP AROUND CONTACT:	ANY CONFIGURATION, SPINE OR FULL WRAP
GRID LINES:	WIDTH, 2 MILS TO 8 MILS; NUMBER 3/Cm to 12/Cm

or silver-chromium. Interconnect materials will be beryllium-copper, Kovar, molybdenum, or silver. (The latter two in a mesh configuration have exhibited superior overall electrical and mechanical behavior in recent welded contact studies).

(e) Current voltage characteristics at 1 AU, 0.010 inch, 28°C. See Figure 4.

(f) Solar cell efficiency as a function of cell thickness between 0.002 and 0.008 inch. See Figure 5.

The following assumptions may be used with regard to the candidate solar cell data:

1. Cells as thin as 0.003-inch can be welded with little or no degradation in performance.
2. Cells as thin as 0.003-inch can be temperature cycled between -190°C and +140°C without incurring damage.
3. The use of alternate cells is not to be considered in the baseline design.
4. The candidate cells described will be available in production quantities in the timeframe necessary to fabricate actual arrays.
5. The fill factor will not change over the temperature range of -100°C to +100°C.
6. Cells thinner than 0.003 inch are not considered practical and will not be used in the baseline design.

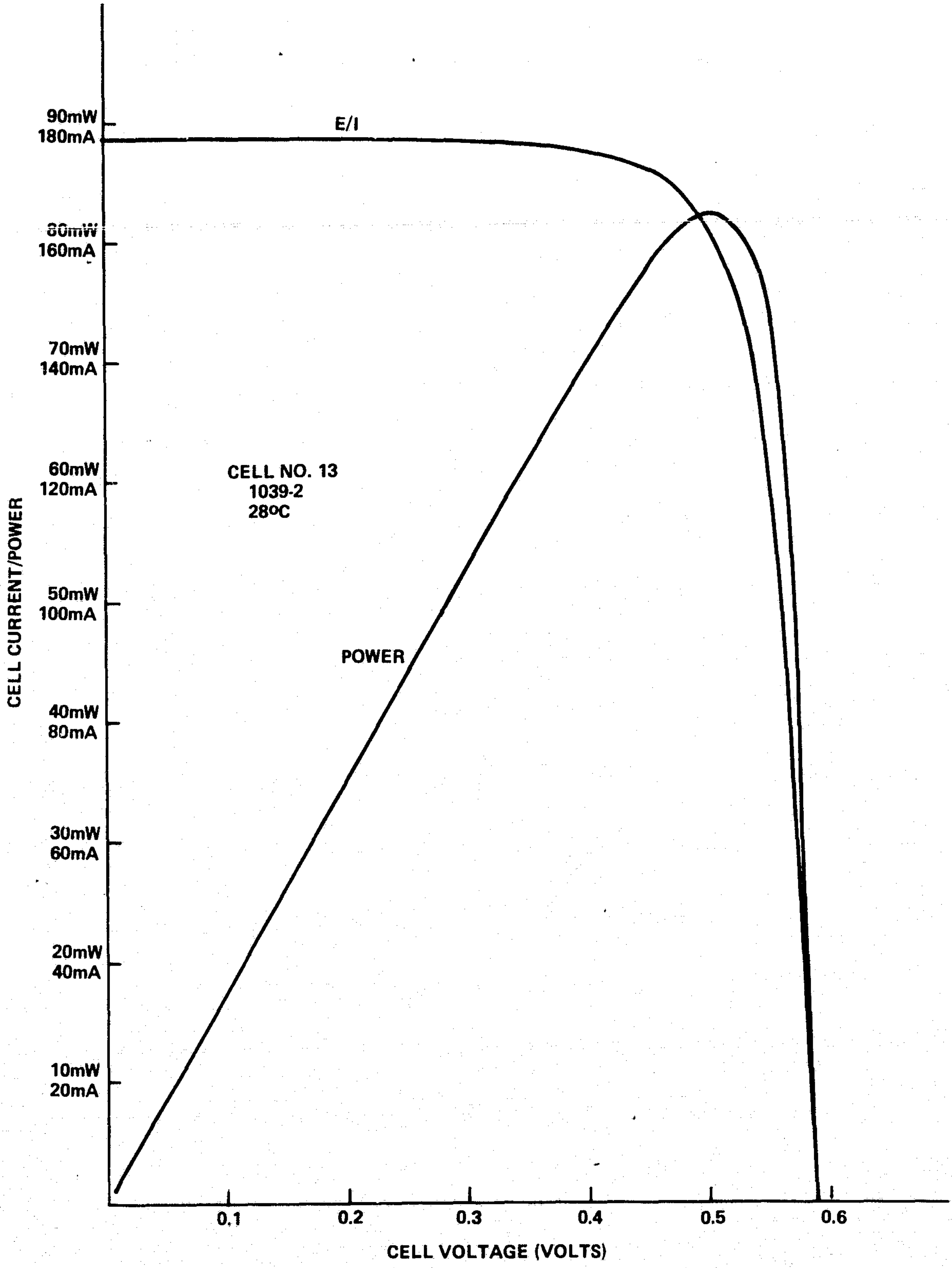


Figure 4 Typical Cell E/I and Power Curves

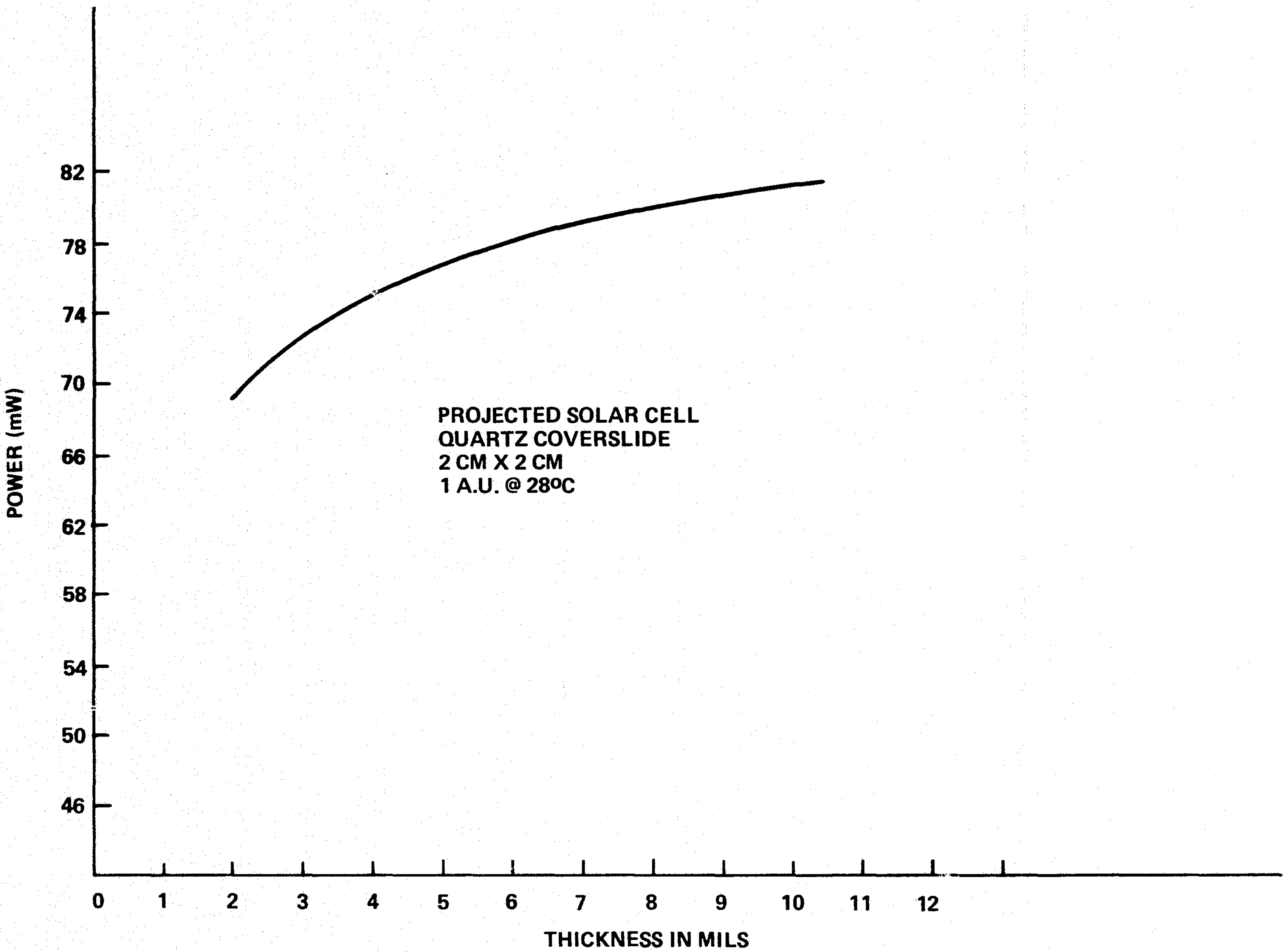


Figure 5 Cell Efficiency vs. Cell Thickness

3.4.6 Solar cell adhesives. A requirement for two separate adhesives can exist in the solar cell area. One requirement shall be for an adhesive used to attach the solar cells to the structures; the second shall be to bond solar cell cover glasses to the cells. The adhesive for bonding cover glasses to solar cells shall be transparent to electromagnetic radiation in wavelengths from 0.4 to 1.0 micron, and shall be resistant to ultraviolet and particulate radiation. The adhesives shall have the following properties:

a) High thermal conductivity
b) Low outgassing in the vacuum environment
c) A modulus of elasticity compatible with the thermal motion of the cells and structure
d) Repairability during the fabrication phase.

3.4.7 Solar cell adhesive thickness tolerance. Solar panel and solar cell installation normally shall require the extensive use of bonding materials. The thickness and area of application of these materials, if used, shall be accurately controlled. The designs and processes shall include control requirements and tolerances that can be maintained in the fabrication shops.

3.4.8 Solar cell tolerances. The control of solar cell processing through the fabrication shops shall be dependent upon the comparison of initial testing and grading to subsequent cell testing during the fabrication sequence. The tolerances set by the design shall be adequate to allow a high yield of good assemblies.

3.4.9 Solar cell connections. The heat required in joining solar cells can cause degradation in cell performance. The solar cell electrical connecting technique shall be compatible with solar cell interconnection methods and shall exhibit accurate temperature control for minimum power loss.

3.4.10 Solar cell installation. The installation of solar cell assemblies onto substrate panels and the assembly of structural component parts shall be accomplished with protective coverings on the operator's hands, or the handling shall be done with suitable mechanical devices. The configuration of these assemblies shall be designed so that the required work can be accomplished while complying with all handling restrictions.

3.4.11 Thermal control coatings. Degradation of thermal control coatings by the ultraviolet and particulate radiation of the flight environment shall be considered.

3.4.12 Bearings and lubricants. In the event bearings and lubricants are required in the solar panel design, the bearing materials shall resist the thermal excursions and particulate radiation of the flight environment. Lubricants shall not degrade; i.e., lose lubricity under flight conditions up to 1095 days, or release any condensing gases, which would case degradation to the spacecraft system. Possible occurrence of cold welding at hard vacuum shall be evaluated.

3.4.13 Part producibility. Configuration and size of parts shall be compatible with normal tooling practices. Very thin foil gage parts shall be capable of being fabricated with reasonable assurance that damage will not occur and that the part can be handled without damage when reasonable precaustions are taken.

3.4.14 Configuration of the solar panel. The configuration of the solar panel shall be designed so that positioning and holding of components and subassemblies can be accomplished to provide support during solar panel assembly.

3.4.15 Repair and replacement. It shall be possible for fabrication personnel to repair or replace any components of the solar panel at any time during the fabrication or ground handling sequence and prior to installation on the spacecraft.

3.5 Mechanical design criteria. The following criteria shall govern the mechanical design of the solar panel.

3.5.1 Strength and deflection requirements. All structures, with minimum material and geometric properties, shall have adequate strength and rigidity to accomplish all requirements. In the fulfillment of the strength and deflection requirements, the worst possible combination of simultaneously applied loads and environmental conditions shall be used to determine limit loads and design loads. Particular attention shall be given to the following.

3.5.1.1 Dynamic loads. During the loads analysis, consideration shall be given to loads induced by the solar panel's elastic and rigid-body response to dynamic excitation in the stowed and/or deployed configuration.

3.5.1.2 Quasi-static loads. The quasi-static loads as a result of vehicles thrust and flight maneuvers shall be 1×10^{-3}g acting upon the deployed solar array.

3.5.1.3 Fatigue considerations. Fatigue shall be considered in the design of structural elements by the avoidance of deleterious residual stresses and stress concentrations in conformity with good design practice. Special attention shall be given to elements subjected to repeated load cycles at high stress levels. Material selection shall include consideration of fatigue characteristics in relation to the design requirements of the structural element.

3.5.1.4 Thermal considerations. Consideration shall be given to deterioration of material properties and to stresses and deformation caused by temperature effects, both prolonged and transient.

3.5.2 Limit load. The limit load shall be the maximum load a structural element is expected to experience during its required functional lifetime, including fabrication, handling, and ground testing. no structural element with minimum material and geometric properties shall yield at limit loads or impair the required functions of the solar panel.

3.5.3 **Design load.** The design load shall be the limit load multiplied by the safety factor. No structural element with minimum material and geometric properties shall experience ultimate stress, failure by instability, or rupture at design load.

3.5.4 **Material properties.** Allowable material properties shall be selected to satisfy the environmental conditions that affect material properties. As a goal, metallic materials shall be in accordance with MIL-HDBK 5.

3.5.5 **Safety factors.** The safety factor is a multiplying factor applied to the limit load to allow for design uncertainties. The following safety factors shall be used as a goal:

a) Structures: 1.25
b) Structural joints, fittings, and brittle material: 1.44.

3.5.6 **Structural qualification test levels.** The environmental levels defined in 3.3 shall be considered as the qualification test levels.

3.5.7 **Structural design.** Simplicity of the analyses and tests shall be considered in the structural design. All structural components shall be amenable to either analytical or experimental demonstration of adequacy.

3.5.8 **Structural nonlinearities.** Nonlinear structural design shall be kept to a minimum and used only when linear behavior is not possible.

3.5.8.1 **Energy dissipating mechanisms.** Where possible, all energy dissipating mechanisms used shall have linear force-velocity relationships over a wide range of frequencies, loads and temperatures.

3.5.8.2 Mechanical backlash. Particular effort shall be made to avoid mechanical backlash in all structural connections.

3.5.9 Separation joint preload. Attachment of any component to another shall provide for sufficient preload so that no physical separation will occur during ultimate load conditions.

3.5.10 Design flexibility. Where practical, the solar panel shall be designed so that additional data and advances of technology may be incorporated at later dates.

3.5.11 Thermal gradients. The solar panel shall be designed to minimize thermal gradients in the plane of the solar panel.

3.5.12 Mechanical integrity. The solar panel shall be designed to prevent the release of loose parts or gases that could damage or impair the function of the solar panel or other spacecraft subsystems.

3.5.13 Margins of safety. Margins of safety are defined with respect to the limit load or the design load as:

$$MS = \frac{* \quad (\text{or} \quad ** \quad)}{\text{limit load (or design load)}} = 1$$

*Load corresponding to yield stress of a structure with minimum geometric and material properties, with consideration of environmental effects on material properties.

**Load corresponding to ultimate stress, instability, or rupture of a structure with minimum geometric and material properties, with consideration of environmental effects on material properties.

3.6 Electrical design criteria. The following criteria shall govern the electrical design of the solar panel.

3.6.1 Solar cell efficiency. The contractor shall establish the power output based on the photovoltaic characteristics of the JPL proposed candidate solar cells and the predicted operating temperature of the solar panel. This design effort shall include the power losses incurred during fabrication, assembly, cabling, and solar panel/spacecraft interfacing considerations.

3.6.2 Electrical insulation. The electrical insulation between the solar cells and the solar panel structure shall provide a maximum dielectric breakdown strength in air, at standard temperatures and pressure conditions, greater than three times the open circuit voltage of the solar panel. Leakage resistance under the test conditions shall be greater than 10^9 ohms per square centimeter of cell area.

3.6.3 Repairability. The solar cell modules shall be constructed, and materials shall be selected, so that any defective cell can be replaced in a fabrication repair area without damage to adjacent cells, electrical insulation, or mounting substrate.

3.6.4 Compatibility of materials. The solar cell stack shall be designed to use only materials that are compatible thermally, mechanically, and electrically with each other, with the space environment, and interface requirements of the solar cells substrate.

3.6.5 Interconnections. The solar cells shall be interconnected, both in parallel and in series by metallic conductors. These conductors shall be designed to minimize both thermal and flexural stresses on the solar cell interconnections. The electrical resistance of the interconnections (including solder, if used) shall not exceed 2 percent of the total series resistance of the solar cells. The joint shall be at least as strong as the bond between the semi-conductor meterial and the ohmic contacts. The joining materials shall exhibit stable physical and electrical characteristics in both space and terrestrial environments.

3.6.6 Magnetic field. Solar cell wiring, interconnecting and structural techniques shall be designed to minimize as far as practical the magnetic field produced by the flow of current in the solar panel.

3.6.7 Electrical conductors. The size and configuration of electrical conductors shall be determined by the following considerations:

a) Low weight
b) Low resistivity
c) Minimum magnetic field
d) Mechanical strength to endure design loads
e) Exterior finish to be resistant to natural and induced environments
f) Process adaptability
g) Redundancy
h) Thermal coefficient of expansion
i) Thermal shock (minimum of 30°C/minute) on the cells
j) Repairability
k) Conductor flexibility.

3.6.8 <u>Conductor insulation.</u> Conductor insulating materials shall be selected on the basis of the following considerations:

a) Mechanical strength
b) Flexibility
c) Dielectric characteristics
d) Ease of forming or fabricating
e) Flight environment considerations
f) Weight.

3.6.9 <u>Electrical terminals.</u> Terminals shall be used to facilitate maintenance, repair, and replacement of electrical components. The following requirements for terminals shall be met:

a) Voltage drop across any terminal shall not exceed 25 millivolts at rated load.

b) The terminals shall withstand 50 cycles of manual mating and unmating without replacement of parts.

c) The terminals shall be accessible for ease of wiring installation and for factory or field checkout.

d) The terminals shall be rigidly attached to primary or secondary structure.

e) The terminals shall have minimum possible weight.

f) Exterior finish of the terminals shall be resistant to both natural and induced environments.

3.6.10 <u>Installation.</u> The installation of wires, terminals, electrical connectors, and busses shall conform to the following requirements:

a) Busses and other wiring shall be installed in order to minimize as far as practical magnetic fields.

b) Installation shall withstand the rigors of normal handling and transportation as well as launch and operational maneuvers.

c) Installation shall be designed to facilitate service and repair activities.

3.6.11 Electrical checkout. Test terminals shall be provided on the solar panel to permit ground testing and checkout prior to launch, in a one-g Earth field, with suitable ground support equipment (GSE).

Designation: E 490 - 73a

Standard Specification for
SOLAR CONSTANT AND AIR MASS ZERO SOLAR SPECTRAL IRRADIANCE[1]

This Standard is issued under the fixed designation E 490; the number immediately following the designation indicates the year of original adoption or, in the case of revision, the year of last revision. A number in parentheses indicates the year of last reapproval.

1. Scope

1.1 This specification defines the solar constant and the zero air mass solar spectral irradiance for use in thermal analysis, thermal balance testing, and other tests of spacecraft and spacecraft components and materials.

1.2 This specification is based upon data from experimental measurements made from high-altitude aircraft, balloons, spacecraft, and the earth's surface. The stated accuracies are based on the estimated accuracies of the measurements, calibrations, and radiometric scales.

2. Applicable Documents

2.1 *ASTM Standards:*

E 349 Definitions of Terms Relating to Space Simulation[2]

3. Definitions of Terms

3.1 *air mass* (optical air mass) (AM)—the ratio of the path length or radiation through the atmosphere (l_m) at any given angle, Z deg, to the sea level path length toward the zenith (l_z).

$$AM = l_m/l_z \cong \sec Z, \text{ for } Z \leq 62 \text{ deg}$$

Symbol: AM1 (air mass one), AM2 (air mass two)

3.2 *air mass zero* (AM0)—the absence of atmospheric attenuation of the solar irradiance at one astronomical unit from the sun.

3.3 *astronomical unit* (AU)—a unit of length defined as the mean distance between the earth and the sun that is, 149 597 890 ± 500 km).

3.4 *irradiance at a point on a surface* (E)—quotient of the radiant flux incident on an element of the surface containing the point, by the area of that element, measured in $W \cdot m^{-2}$.

3.5 *irradiance, spectral* (E_λ)—the irradiance per unit wavelength interval at a specific wavelength, or as a function of wavelength measured in $W \cdot m^{-2} \cdot \mu m^{-1}$.

3.6 *integrated irradiance*—spectral irradiance integrated over a specific wavelength interval from λ_1 to λ_2, measured in $W \cdot m^{-2}$. Symbol: $E_{\lambda_1 - \lambda_2} = \int_{\lambda_1}^{\lambda_2} E_\lambda d\lambda$

3.7 Additional definitions will be found in Definitions E 349.

4. Solar Constant

4.1 The solar constant, based on the average of the values shown in Table 1, is 1353 $W \cdot m^{-2}$. The estimated error is ±21 $W \cdot m^{-2}$.

4.2 Table 2 summarizes the results in different units, and Table 3 presents the total solar irradiance at various planetary distances from the sun.

5. Solar Spectral Irradiance (Air Mass Zero)

5.1 The zero air mass solar spectral irradiance is based on data from the NASA 711 research aircraft experiments (**1,2,3**)[3] (see Table 4) with additions and revisions based on other recent measurements (**16**). Previously compiled solar spectral irradiances were based on ground-based measurements (**17** to **25**) and some measurements from rockets (**26**). Spectral irradiance data from the NASA Ames Research Center (**27**) were not included because of

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[1] This specification is under the jurisdiction of ASTM Committee E-21 on Space Simulation.

Current edition approved Sept. 27, 1973 and Dec. 27, 1973. Published January 1974.

[2] *1974 Annual Book of ASTM Standards*, Part 41

[3] The boldface numbers in parentheses refer to the list of references at the end of this specification.

calibration uncertainties. Further discussion on the methods of calculation and historical information can be found in Refs (**3,16,28** to **31**).

5.2 Table 5 presents the solar spectral irradiance in tabular form for the range from 0.115 to 1000 μm. The first column gives the wavelength (λ) in μm; the second gives the spectral irradiance (E_λ) at λ in $W \cdot m^{-2} \cdot \mu m^{-1}$; the third gives the total irradiance for the range from 0 to λ ($E_{0-\lambda}$) in $W \cdot m^{-2}$; and the fourth gives the percentage of the solar constant associated with wavelengths shorter than λ ($D_{0-\lambda}$).

5.3 Table 6 presents an abridged version of Table 5. Figure 1 plots the Standard Solar Spectral Irradiance.

5.4 The irradiance in the range from 0 to 0.115 μm (nearly 0.0025 $W \cdot m^{-2}$) is based on Hinteregger's results (**32**). In the 0.14 to 0.20-μm range, the values are based on Naval Research Laboratory data (**17, 26**), which have been adjusted downward because of data by Heath (**33**) and Parkinson and Reeves (**34**). In the range from 0.20 to 0.30 μm, the values of the Goddard Space Flight Center curve have been retained because of confirming Nimbus satellite data (**33**). The Epply-JPL data were used for revision in the range from 0.3 to 0.7 μm (**9** to **13**). The 20 to 1000-μm range (**9** to **13, 16**) irradiances were computed from the combined data on the brightness temperature of the sun from many different authors as quoted by Shimabukoro and Stacey (**35**).

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TABLE 1 Solar Constant

Platform	Detector	Year	Solar Constant, $W \cdot m^{-2}$	Ref
NASA 711 aircraft	Hy-Cal pyrheliometer	1967	1358	1,2,3
NASA 711 aircraft	Angström 7635	1967	1349	1,2,3
NASA 711 aircraft	Angström 6618	1967	1343	1,2,3
NASA 711 aircraft	cone radiometer	1967	1358	1,2,3
Soviet balloon	U. of Leningrad actinometer	1961–1968	1353	4,5
U. of Denver balloon	Eppley pyrheliometer	1969	1338	6,7
Eppley-JPL high-altitude aircraft	Eppley pyrheliometer	1966–1968	1360	8 to 13
Mariner VI and VIII spacecraft	cavity radiometer	1969	1353	14
JPL balloon	cavity radiometer	1968–1969	1368	15
Average estimated error			1353 ± 21	

TABLE 2 Solar Constant Conversion Factors

Solar constant	= 1353 $W \cdot m^{-2}$ ($\pm$21 $W \cdot m^{-2}$) [preferred unit]
	= 0.1353 $W \cdot cm^{-2}$
	= 135.3 $mW \cdot cm^{-2}$
	= 1.353 $\times 10^{6}$ $erg \cdot cm^{-2} \cdot s^{-1}$
	= 125.7 $W \cdot ft^{-2}$
	= 1.940 $cal \cdot cm^{-2} \cdot min^{-1}$ ($\pm$0.03 $cal \cdot cm^{-2} \cdot min^{-1}$)
	= 0.0323 $cal \cdot cm^{-2} \cdot s^{-1}$
	= 429.2 $Btu \cdot ft^{-2} \cdot h^{-1}$
	= 0.119 $Btu \cdot ft^{-2} \cdot s^{-1}$
	= 1.937 $Langleys \cdot min^{-1}$

The calorie is the thermochemical calorie-gram and is defined as 4.1840 absolute joules. The Btu is the thermochemical British thermal unit and is defined by the relationship: 1 Btu (thermochemical)/(°F × lb) = 1 cal·g (thermochemical)/(°C × g).

The Langley, however, is defined in terms of the older thermal unit the calorie·g (mean), that is, 1 Langley = 1 cal·g (mean)·cm^{-2}; 1 cal·g (mean) = 4.19002 J.

TABLE 3 Solar Irradiance at the Planets

Planet	Solar Irradiance, $W \cdot m^{-2}$		
	Mean	Perihelion	Aphelion
Mercury	9029.0	14309.0	6211.0
Venus	2586.0	2621.0	2551.0
Earth	1353.0	1399.0	1309.0
Mars	583.0	709.0	487.0
Jupiter	50.0	55.2	45.5
Saturn	14.9	16.6	13.4
Uranus	3.68	4.07	3.34
Neptune	1.496	1.500	1.493
Pluto	0.870	1.556	0.555

TABLE 4 Spectral Irradiance Instruments On Board the NASA 711 Galileo Research Aircraft, Used for Obtaining the GSFC Curve of Solar Spectral Irradiance (Refs 1, 2, 3)

Instrument	Energy Detector	Type of Instrument	Aircraft Window Material	Wavelength Range, μm
Perkin-Elmer monochromator	1P28 tube, thermocouple	LiF prism	sapphire	0.3–0.7 0.7–4
Leiss monochromator	EMI 9558QA, PbS cell	quartz double prism	Dynasil quartz	0.3–0.7 0.7–1.6
Filter radiometer	phototube	dielectric thin films	Dynasil quartz	0.3–1.2
P-4 interferometer	1P28 or R136 PbS cell	Soleil prism	Infrasil quartz	0.3–0.7 0.7–2.5
I-4 interferometer	thermistor bolometer	Michelson mirror	Irtran 4	2.6–15

TABLE 5 Solar Spectral Irradiance—Standard Curve

λ = wavelength, μm,
E_λ = solar spectral irradiance averaged over small bandwidth centered at λ, $W \cdot m^{-2} \cdot \mu m^{-1}$
$E_{0-\lambda}$ = integrated solar irradiance in the wavelength range 0 to λ, in $W \cdot m^{-2}$,
$D_{0-\lambda}$ = percentage of solar constant associated with wavelengths shorter than λ, and
solar constant = 1353 $W \cdot m^{-2}$

NOTE—Lines indicate change in wavelength interval of integration.

λ	E_λ	$E_{0-\lambda}$	$D_{0-\lambda}$	λ	E_λ	$E_{0-\lambda}$	$D_{0-\lambda}$	λ	E_λ	$E_{0-\lambda}$	$D_{0-\lambda}$
.115	.007	.0025	.0001	.510	1882	324.926	24.015	1.55	267	1186.109	87.665
.120	.900	.0048	.0003	.515	1833	334.214	24.701	1.60	245	1198.909	88.611
.125	.007	.0070	.0005	.520	1833	343.379	25.379	1.65	223	1210.609	89.475
.130	.007	.0071	.0005	.525	1852	352.591	26.059	1.70	202	1221.234	90.261
.140	.030	.0073	.0005	.530	1842	361.826	26.742	1.75	180	1230.784	90.967
.150	.070	.0078	.0005	.535	1818	370.976	27.418	1.80	159	1239.259	91.593
.160	.230	.0093	.0006	.540	1783	379.979	28.084	1.85	142	1246.784	92.1[illegible]9
.170	.630	.0136	.0010	.545	1754	388.821	28.737	1.90	126	1253.484	92.644
.180	1.250	.0230	.0016	.550	1725	397.519	29.380	1.95	114	1259.484	93.088
.190	2.710	.0428	.0031	.555	1720	406.131	30.017	2.00	103	1264.909	93.489
.200	10.7	.1098	.0081	.560	1695	414.669	30.648	2.1	90	1274.559	94.2024
.210	22.9	.2778	.0205	.565	1705	423.169	31.276	2.2	79	1283.009	94.8269
.220	57.5	.6798	.0502	.570	1712	431.711	31.907	2.3	69	1290.409	95.3739
.225	64.9	.9858	.0728	.575	1719	440.289	32.541	2.4	62	1296.959	95.8580
.230	66.7	1.3148	.0971	.580	1715	448.874	33.176	2.5	55	1302.809	96.2903
.235	59.3	1.6298	.1204	.585	1712	457.441	33.809	2.6	48	1307.959	96.6710
.240	63.0	1.9356	.1430	.590	1700	465.971	34.439	2.7	43	1312.509	97.0073
.245	72.3	2.2738	.1680	.595	1682	474.426	35.064	2.8	39	1316.609	97.3103
.250	70.4	2.6306	.1944	.600	1666	482.796	35.683	2.9	35	1320.309	97.5838
.255	104.0	3.0666	.2266	.605	1647	491.079	36.295	3.0	31	1323.609	97.8277
.260	130	3.6516	.269	.61	1635	499.284	36.902	3.1	26.0	1326.459	98.0383
.265	185	4.4391	.328	.62	1602	515.469	38.098	3.2	22.6	1328.889	98.2179
.270	232	5.4816	.405	.63	1570	531.329	39.270	3.3	19.2	1330.979	98.3724
.275	204	6.5716	.485	.64	1544	546.899	40.421	3.4	16.6	1332.769	98.5047
.280	222	7.6366	.564	.65	1511	562.174	41.550	3.5	14.6	1334.329	98.6200
.285	315	8.9791	.663	.66	1486	577.159	42.657	3.6	13.5	1335.734	98.7238
.290	482	10.9716	.810	.67	1456	591.869	43.744	3.7	12.3	1337.024	98.8192
.295	584	13.6366	1.007	.68	1427	606.284	44.810	3.8	11.1	1338.194	98.9056
.300	514	16.3816	1.210	.69	1402	620.429	45.855	3.9	10.3	1339.264	98.9847
.305	603	19.1741	1.417	.70	1369	634.284	46.879	4.0	9.5	1340.254	99.0579
.310	689	22.4041	1.655	.71	1344	647.849	47.882	4.1	8.70	1341.1641	99.12521
.315	764	26.0366	1.924	.72	1314	661.139	48.864	4.2	7.80	1341.9891	99.18618
.320	830	30.0216	2.218	.73	1290	674.159	49.826	4.3	7.10	1342.7341	99.24124
.325	9[illegible]	34.5341	2.552	.74	1260	686.909	50.769	4.4	6.50	1343.4141	99.29150
.330	1059	39.6191	2.928	.75	1235	699.384	51.691	4.5	5.92	1344.0351	99.33740
.335	1081	44.9691	3.323	.76	1211	711.614	52.595	4.6	5.35	1344.5986	99.37905
.340	1074	50.3566	3.721	.77	1185	723.594	53.480	4.7	4.86	1345.1091	99.41678
.345	1069	55.7141	4.117	.78	1159	735.314	54.346	4.8	4.47	1345.5757	99.45127
.350	1093	61.1191	4.517	.79	1134	746.779	55.194	4.9	4.11	1346.0049	99.48299
.355	1083	66.5591	4.919	.80	1109	757.994	56.023	5.0	3.79	1346.3999	99.51219
.360	1068	71.9366	5.316	.81	1085	768.966	56.834	6	1.8200	1349.2049	99.71950
.365	1132	77.4366	5.723	.82	1060	779.694	57.627	7	.9900	1350.6099	99.82335
.370	1181	83.2191	6.150	.83	1036	790.174	58.401	8	.5850	1351.3974	99.88155
.375	1157	89.0641	6.582	.84	1013	800.419	59.158	9	.3670	1351.8734	99.91673
.380	1120	94.7566	7.003	.85	990	810.434	59.899	10	.2410	1352.1774	99.93920
.385	1098	100.3016	7.413	.86	968	820.224	60.622	11	.1650	1352.3804	99.95420
.390	1098	105.7916	7.819	.87	947	829.799	61.330	12	.1170	1352.5214	99.96462
.395	1189	111.5091	8.241	.88	926	839.164	62.022	13	.0851	1352.6224	99.97209
.400	1429	118.0541	8.725	.89	908	848.334	62.700	14	.0634	1352.6967	99.97758
.405	1644	125.7366	9.293	.90	891	857.329	63.365	15	.0481	1352.7524	99.98170
.410	1751	134.224	9.920	.91	880	866.184	64.019	16	.037100	1352.7950	99.98485
.415	1774	143.036	10.571	.92	869	874.929	64.665	17	.029100	1352.8281	99.98730
.420	1747	151.839	11.222	.93	858	883.564	65.304	18	.023100	1352.8542	99.98923
.425	1693	160.439	11.858	.94	847	892.089	65.934	19	.018600	1352.8751	99.99077
.430	1639	168.769	12.473	.95	837	900.509	66.556	20	.015200	1352.8920	99.99202
.435	1663	177.024	13.083	.96	820	908.794	67.168	25	.006170	1352.9454	99.99596
.440	1810	185.706	13.725	.97	803	916.909	67.768	30	.002970	1352.9683	99.99765
.445	1922	195.036	14.415	.98	785	924.849	68.355	35	.001600	1352.9797	99.99850
.450	2006	204.856	15.140	.99	767	932.609	68.928	40	.000942	1352.9860	99.99897
.455	2057	215.014	15.891	1.00	748	940.184	69.488	50	.000391	1352.9927	99.99946
.460	2066	225.321	16.653	1.05	668	975.584	72.105	60	.00019000	1352.9956	99.99967
.465	2048	235.606	17.413	1.10	593	1007.109	74.435	80	.00006160	1352.9981	99.99986
.470	2033	245.809	18.167	1.15	535	1035.309	76.519	100	.00002570	1352.9990	99.99992
.475	2044	256.001	18.921	1.20	485	1060.809	78.404	120	.00001260	1352.9994	99.99995
.480	2074	266.296	19.681	1.25	438	1083.884	80.109	150	.00000523	1352.9997	99.99997
.485	1976	276.421	20.430	1.30	397	1104.759	81.652	200	.00000169	1352.9998	99.99999
.490	1950	286.236	21.155	1.35	358	1123.634	83.047	250	.00000070	1352.9999	99.99999
.495	1960	296.011	21.878	1.40	337	1141.209	84.331	300	.00000034	1352.9999	99.99999
.500	1942	305.766	22.599	1.45	312	1157.234	85.530	400	.00000011	1352.9999	99.99999
.505	1920	315.421	23.312	1.50	288	1172.234	86.639	1000	.00000000	1353.0000	100.00000

TABLE 6 Solar Spectral Irradiance—Standard Curve, Abridged Version

λ = wavelength, μm,
E_λ = solar spectral irradiance averaged over small bandwidth centered at λ, $W \cdot m^{-2} \cdot \mu m^{-1}$,
$D_{o-\lambda}$ = percentage of the solar constant associated with wavelengths shorter than λ, and
solar constant = 1353 $W \cdot m^{-2}$

λ	E_λ	$D_{o-\lambda}$	λ	E_λ	$D_{o-\lambda}$	λ	E_λ	$D_{o-\lambda}$
0.115	.007	1×10^{-4}	0.43	1639	12.47	0.90	891	63.37
0.14	.03	5×10^{-4}	0.44	1810	13.73	1.00	748	69.49
0.16	.23	6×10^{-4}	0.45	2006	15.14	1.2	485	78.40
0.18	1.25	1.6×10^{-3}	0.46	2066	16.65	1.4	337	84.33
0.20	10.7	8.1×10^{-3}	0.47	2033	18.17	1.6	245	88.61
0.22	57.5	0.05	0.48	2074	19.68	1.8	159	91.59
0.23	66.7	0.10	0.49	1950	21.15	2.0	103	93.49
0.24	63.0	0.14	0.50	1942	22.60	2.2	79	94.83
0.25	70.9	0.19	0.51	1882	24.01	2.4	62	95.86
0.26	130	0.27	0.52	1833	25.38	2.6	48	96.67
0.27	232	0.41	0.53	1842	26.74	2.8	39	97.31
0.28	222	0.56	0.54	1783	28.08	3.0	31	97.83
0.29	482	0.81	0.55	1725	29.38	3.2	22.6	98.22
0.30	514	1.21	0.56	1695	30.65	3.4	16.6	98.50
0.31	689	1.66	0.57	1712	31.91	3.6	13.5	98.72
0.32	830	2.22	0.58	1715	33.18	3.8	11.1	98.91
0.33	1059	2.93	0.59	1700	34.44	4.0	9.5	99.06
0.34	1074	3.72	0.60	1666	35.68	4.5	5.9	99.34
0.35	1093	4.52	0.62	1602	38.10	5.0	3.8	99.51
0.36	1068	5.32	0.64	1544	40.42	6.0	1.8	99.72
0.37	1181	6.15	0.66	1486	42.66	7.0	1.0	99.82
0.38	1120	7.00	0.68	1427	44.81	8.0	.59	99.88
0.39	1098	7.82	0.70	1369	46.88	10.0	.24	99.94
0.40	1429	8.73	0.72	1314	48.86	15.0	4.8×10^{-2}	99.98
0.41	1751	9.92	0.75	1235	51.69	20.0	1.5×10^{-2}	99.99
0.42	1747	11.22	0.80	1109	56.02	50.0	3.9×10^{-4}	100.00

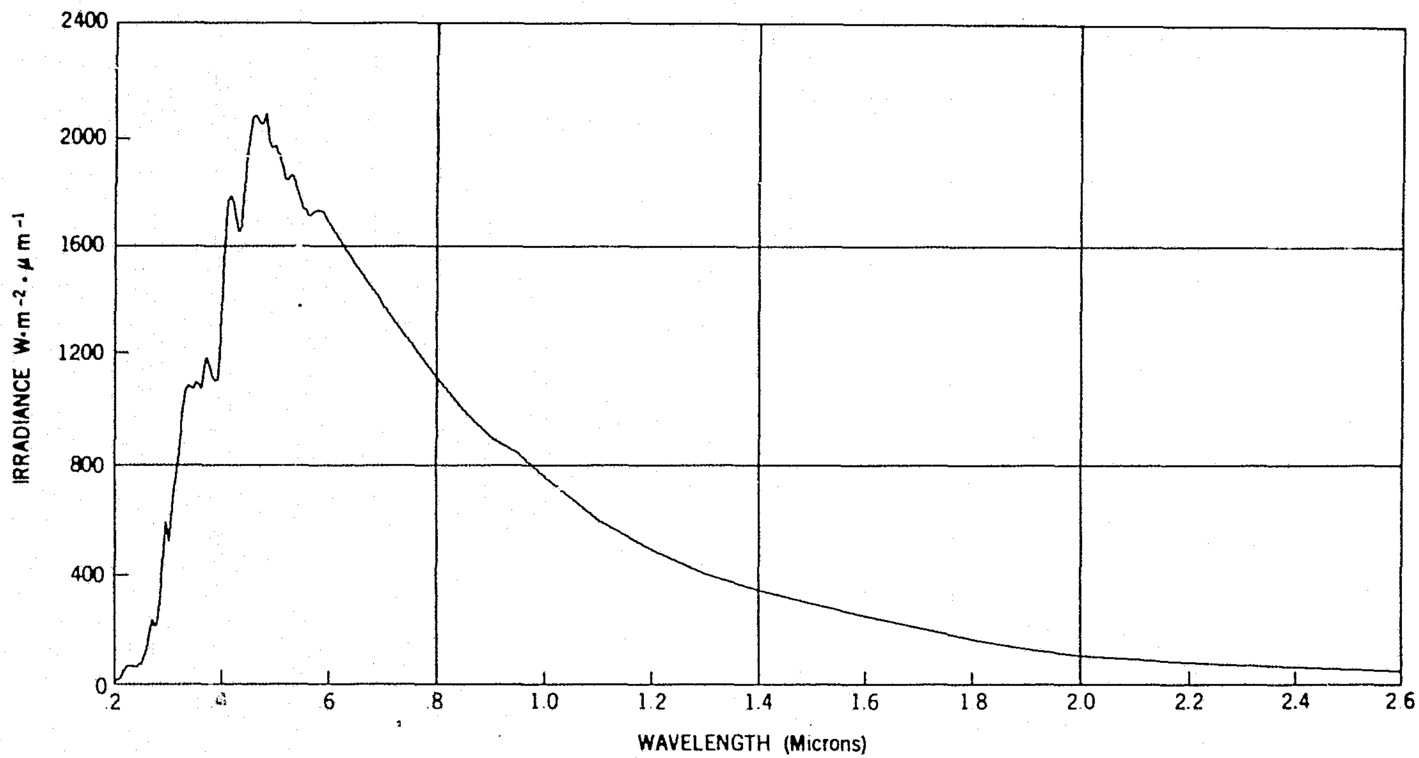

FIG. 1 Solar Spectral Irradiance.

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